Space Transfer Concepts and Analysis for Exploration Missions

Implementation Plan and Element Description Document (draft final)
Volume 2: Cryo/Aerobrake Vehicle

March 8, 1991

Boeing Aerospace and Electronics
Huntsville, Alabama
NASA Contract NAS8-37857
Space Transfer Concepts and Analyses for Exploration Missions

NASA Contract NAS8-37857

Cryogenic/Aerobrake Vehicle Implementation Plan and Element Description Document

Boeing Aerospace and Electronics
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Date 3/15/91

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Space Transfer Concepts and Analyses for Exploration Missions

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Boeing Aerospace and Electronics
Huntsville, Alabama

Documentation Set:
D615-10026-1 IP and ED Volume 1: Major Trades, Books 1 and 2
D615-10026-2 IP and ED Volume 2: Cryogenic/Aerobrake Vehicle
D615-10026-3 IP and ED Volume 3: Nuclear Thermal Rocket Vehicle
D615-10026-4 IP and ED Volume 4: Solar Electric Propulsion Vehicle
D615-10026-5 IP and ED Volume 5: Nuclear Electric Propulsion Vehicle
D615-10026-6 IP and ED Volume 6: Lunar Systems
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# Implementation Plan and Element Description Document

## Cryogenic/Aerobrake

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<tr>
<td>ACRV</td>
<td>Advanced crew recovery vehicle</td>
</tr>
<tr>
<td>ACS</td>
<td>Attitude control system</td>
</tr>
<tr>
<td>AFE</td>
<td>Aerobrake Flight Experiment</td>
</tr>
<tr>
<td>A&amp;I</td>
<td>Attachment and integration</td>
</tr>
<tr>
<td>AI</td>
<td>Aluminum</td>
</tr>
<tr>
<td>ALARA</td>
<td>As low as reasonably achievable</td>
</tr>
<tr>
<td>ALS</td>
<td>Advanced Launch System</td>
</tr>
<tr>
<td>ALSPE</td>
<td>Anomalously large solar proton event</td>
</tr>
<tr>
<td>am</td>
<td>Atomic mass (unit)</td>
</tr>
<tr>
<td>AR</td>
<td>Area ratio</td>
</tr>
<tr>
<td>ARGPER</td>
<td>Argument of perigee</td>
</tr>
<tr>
<td>ARS</td>
<td>Atmospheric revitalization system</td>
</tr>
<tr>
<td>art-g</td>
<td>Artificial gravity</td>
</tr>
<tr>
<td>asc</td>
<td>Ascent</td>
</tr>
<tr>
<td>ASE</td>
<td>Advanced space engine</td>
</tr>
<tr>
<td>AU</td>
<td>Astronomical Unit (=149.6 million km)</td>
</tr>
<tr>
<td>BIT</td>
<td>Built-in test</td>
</tr>
<tr>
<td>BITE</td>
<td>Built-in test equipment</td>
</tr>
<tr>
<td>BLAP</td>
<td>Boundary Layer Analysis Program</td>
</tr>
<tr>
<td>BFO</td>
<td>Blood-forming organs</td>
</tr>
<tr>
<td>C</td>
<td>Degrees Celsius</td>
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<tr>
<td>CAB</td>
<td>Cryogenic/aerobrake</td>
</tr>
<tr>
<td>CAD/CAM</td>
<td>Computer-aided design/computer-aided manufacturing</td>
</tr>
<tr>
<td>CAP</td>
<td>Cryogenic all-propulsive</td>
</tr>
<tr>
<td>Cd</td>
<td>Drag coefficient</td>
</tr>
<tr>
<td>CELSS</td>
<td>Closed Environmental Life Support System</td>
</tr>
<tr>
<td>CHC</td>
<td>Crew health care</td>
</tr>
<tr>
<td>CG</td>
<td>Center of gravity</td>
</tr>
<tr>
<td>CL</td>
<td>Lift coefficient</td>
</tr>
<tr>
<td>cm</td>
<td>Centimeter = 0.01 meter</td>
</tr>
<tr>
<td>ctm</td>
<td>Crew module</td>
</tr>
<tr>
<td>CM</td>
<td>Center of mass</td>
</tr>
<tr>
<td>c/o</td>
<td>Check out</td>
</tr>
<tr>
<td>C of F</td>
<td>Cost of facilities</td>
</tr>
<tr>
<td>conj</td>
<td>Conjunction</td>
</tr>
<tr>
<td>COSPAR</td>
<td>Committee on Space Research of the International Council of Scientific Unions</td>
</tr>
<tr>
<td>CO2</td>
<td>Carbon dioxide</td>
</tr>
<tr>
<td>Cryo</td>
<td>Cryogenic</td>
</tr>
<tr>
<td>C3</td>
<td>Hyperbolic excess velocity squared (in km²/s²)</td>
</tr>
<tr>
<td>d</td>
<td>days</td>
</tr>
<tr>
<td>DDT&amp;E</td>
<td>Design, development, testing, and evaluation</td>
</tr>
<tr>
<td>DE</td>
<td>Dose equivalent</td>
</tr>
<tr>
<td>deg</td>
<td>Degrees</td>
</tr>
<tr>
<td>desc</td>
<td>Descent</td>
</tr>
<tr>
<td>DMS</td>
<td>Data management system</td>
</tr>
<tr>
<td>dV</td>
<td>Velocity change (ΔV)</td>
</tr>
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</table>
Earth arrival
Earth arrival
Modulus of elasticity in compression
Earth crew capture vehicle
Element control work station
Environment control and life support system
Electric propulsion
European Space Agency
Engine start opportunity
External Tank
Earth-to-orbit
Extra-vehicular activity
Circulation efficiency factor
Fire Detection and Differentiation
Life support weight factor
Specific floor count factor
Specific floor area factor
Aerobrake integration factor
Specific length factor
Normalized spatial unit count factor
Path options factor
Useful perimeter factor
Parts count factor
Proximity convenience factor
Plan aspect ratio factor
Section aspect ratio factor
Flight support equipment
Vault factor
Safe-haven split factor
Spatial unit number factor
Volume range factor
Fiscal Year 1988 (=October 1, 1987 to September 30, 1988. Similarly for other years)
Acceleration in Earth gravities (=acceleration/9.80665 m/s²)
Gas core nuclear rocket
Galactic cosmic rays
Geosynchronous Earth Orbit
Gaseous nitrogen
Guidance, navigation, and control
Global Positioning System
Gray (SI unit of absorbed radiation energy = 10⁴ erg/gm)
Habitation
High Density
Human Exploration Initiative (obsolete for SEI)
Heavy lift launch vehicle
Hours
Hygiene water
High atomic number and energy particle
Hydrogen
Water
ICRP
IMLEO
in.
inb
IP&ED
IR&D
Isp
ISRU
JEM
JSC
k
keV
kg
klb
klbf
km
KM
KM/Sec
KM/SEC
ksi
L/D
LD
LDM
LEO
LET
LEV
LEVCM
Level II
LH2
LiOH
LLO
LM
LOR
LOX
LS
LTV
LTVCM
L2
m
[MarsGram
[MARSIN
MASE
MAV
M/CpA
MCRV
me
MEOP
MeV
International Commission on Radiation Protection
Initial mass in low Earth orbit
Inches
Inbound
Implementation Plan and Element Description
Indepandent research and development
Specific impulse (=thrust/mass flow rate)
In-situ resource utilization
Japan Experiment Module (of SSF)
Johnson Space Center
Kilopounds (thousands of pounds. Conversion to SI units=4448 N/klb)
Kilopound force
Kilometers
Kilometers
Kilometers per second
Kilometers per second
Kilopounds per square inch
Lift-to-drag ratio
Low density
Long duration mission
Low Earth orbit
Linear energy transfer
Lunar excursion vehicle
Lunar excursion vehicle crew module
Space Exploration Initiative project office, Johnson Space Center
Liquid hydrogen
Lithium hydroxide
Low Lunar orbit
Lunar Module
Lunar orbit rendezvous
Liquid oxygen
Lunar surface
Lunar transfer vehicle
Lunar transfer vehicle crew module
Lagrange point 2. A point behind the Moon as seen from the Earth which
has the same orbital period as the moon.
Meters
Western Union interplanetary telegram
Martian pornography
Mission analysis and systems engineering (same as Level II q.v.)
Mars ascent vehicle
Ballistic coefficient (mass / drag coefficient times area)
Modified crew recovery vehicle
Mass of electron
Maximum expected operating pressure
Million electron volt
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<thead>
<tr>
<th>Abbreviation</th>
<th>Description</th>
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<tr>
<td>MEV</td>
<td>Mars excursion vehicle</td>
</tr>
<tr>
<td>MLI</td>
<td>Multi-layer insulation</td>
</tr>
<tr>
<td>mm</td>
<td>Millimeter (=0.001 meter)</td>
</tr>
<tr>
<td>MMH</td>
<td>Monomethylhydrazine</td>
</tr>
<tr>
<td>MMV</td>
<td>Manned Mars vehicle</td>
</tr>
<tr>
<td>MOC</td>
<td>Mars orbit capture</td>
</tr>
<tr>
<td>MOI</td>
<td>Mars orbit insertion</td>
</tr>
<tr>
<td>mod</td>
<td>Module</td>
</tr>
<tr>
<td>M&amp;P</td>
<td>Materials and processes</td>
</tr>
<tr>
<td>MPS</td>
<td>Main propulsion system</td>
</tr>
<tr>
<td>MR</td>
<td>Mixture ratio</td>
</tr>
<tr>
<td>m/sec</td>
<td>Meters per second</td>
</tr>
<tr>
<td>MSFC</td>
<td>Marshall Space Flight Center</td>
</tr>
<tr>
<td>Msi</td>
<td>Million pounds per square inch</td>
</tr>
<tr>
<td>mt</td>
<td>Metric tons (thousands of kilograms)</td>
</tr>
<tr>
<td>mT</td>
<td>Metric tons</td>
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<tr>
<td>MTBF</td>
<td>Mean time between failures</td>
</tr>
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<td>MTV</td>
<td>Mars transfer vehicle</td>
</tr>
<tr>
<td>MWe</td>
<td>Megawatts electric</td>
</tr>
<tr>
<td>m³</td>
<td>Cubic Meters</td>
</tr>
<tr>
<td>N</td>
<td>Newton. Kilogram-meters per second squared</td>
</tr>
<tr>
<td>n/a</td>
<td>Not applicable</td>
</tr>
<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
</tr>
<tr>
<td>NCRP</td>
<td>National Council on Radiation Protection</td>
</tr>
<tr>
<td>NEP</td>
<td>Nuclear-electric propulsion</td>
</tr>
<tr>
<td>NERVA</td>
<td>Nuclear engine for rocket vehicle application</td>
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<tr>
<td>NSO</td>
<td>Nuclear safe orbit</td>
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<tr>
<td>NTR</td>
<td>Nuclear thermal rocket</td>
</tr>
<tr>
<td>N2O4</td>
<td>Nitrogen tetroxide</td>
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<tr>
<td>OSE</td>
<td>Orbital support equipment</td>
</tr>
<tr>
<td>OTIS</td>
<td>Optimal Trajectories by Implicit Simulation program</td>
</tr>
<tr>
<td>outb</td>
<td>Outbound</td>
</tr>
<tr>
<td>O2</td>
<td>Oxygen</td>
</tr>
<tr>
<td>PBR</td>
<td>Particle bed reactor</td>
</tr>
<tr>
<td>Pc</td>
<td>Chamber pressure</td>
</tr>
<tr>
<td>PEEK</td>
<td>Polyether-ether ketone</td>
</tr>
<tr>
<td>PEGA</td>
<td>Powered Earth gravity assist</td>
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<td>P/L</td>
<td>Payload</td>
</tr>
<tr>
<td>POTV</td>
<td>Personnel orbital transfer vehicle</td>
</tr>
<tr>
<td>pot w</td>
<td>Potable water</td>
</tr>
<tr>
<td>PPU</td>
<td>Power processing unit</td>
</tr>
<tr>
<td>prop</td>
<td>Propellant</td>
</tr>
<tr>
<td>psi</td>
<td>Pounds per square inch</td>
</tr>
<tr>
<td>PV</td>
<td>Photovoltaic</td>
</tr>
<tr>
<td>Q</td>
<td>Heat flux (Joules per square centimeter)</td>
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<tr>
<td>Q</td>
<td>Radiation quality factor</td>
</tr>
<tr>
<td>RAAN</td>
<td>Right ascension of ascending node</td>
</tr>
<tr>
<td>RCS</td>
<td>Reaction control system</td>
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<td>Abbreviation</td>
<td>Description</td>
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</tr>
<tr>
<td>Re</td>
<td>Reynolds number</td>
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<td>RF</td>
<td>Radio frequency</td>
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<td>RMLEO</td>
<td>Resupply mass in low Earth orbit</td>
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<td>RPM</td>
<td>Revolutions per minute</td>
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<td>RWA</td>
<td>Relative wind angle</td>
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<tr>
<td>R&amp;D</td>
<td>Research and Development</td>
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<tr>
<td>Rendezvous and dock</td>
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<tr>
<td>SAA</td>
<td>South Atlantic Anomaly</td>
</tr>
<tr>
<td>SAIC</td>
<td>Science Applications International Corporation</td>
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<tr>
<td>SEI</td>
<td>Space Exploration Initiative</td>
</tr>
<tr>
<td>SEP</td>
<td>Solar-electric propulsion</td>
</tr>
<tr>
<td>SI</td>
<td>International system of units (metric system)</td>
</tr>
<tr>
<td>SiC</td>
<td>Silicon carbide</td>
</tr>
<tr>
<td>SMA</td>
<td>Semimajor axis</td>
</tr>
<tr>
<td>sol</td>
<td>Solar day (24.6 hours for Mars)</td>
</tr>
<tr>
<td>SPE</td>
<td>Solar proton events</td>
</tr>
<tr>
<td>SRB</td>
<td>Solid Rocket Booster</td>
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<td>SSF</td>
<td>Space Station Freedom</td>
</tr>
<tr>
<td>SSME</td>
<td>Space Shuttle Main Engine</td>
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<td>STCAEM</td>
<td>Space Transfer Concepts and Analysis for Exploration Missions</td>
</tr>
<tr>
<td>stg</td>
<td>Stage</td>
</tr>
<tr>
<td>surf</td>
<td>Surface</td>
</tr>
<tr>
<td>Sv</td>
<td>Sievert (SI unit of dose equivalent = Gy x Q)</td>
</tr>
<tr>
<td>S1</td>
<td>Distance along aerobrake surface forward of the stagnation point</td>
</tr>
<tr>
<td>S2</td>
<td>Distance along aerobrake surface aft of the stagnation point</td>
</tr>
<tr>
<td>S3</td>
<td>Distance along aerobrake surface starboard of the stagnation point</td>
</tr>
<tr>
<td>t.</td>
<td>Metric tons (1000kg)</td>
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<tr>
<td>TBD</td>
<td>To be determined</td>
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<tr>
<td>Tc</td>
<td>Chamber temperature</td>
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<tr>
<td>TCS</td>
<td>Thermal control system</td>
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<tr>
<td>TEI</td>
<td>Trans-Earth injection</td>
</tr>
<tr>
<td>TEL</td>
<td>Trans-Earth injection stage</td>
</tr>
<tr>
<td>t.f.</td>
<td>Tank weight factor</td>
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<tr>
<td>THC</td>
<td>Temperature and humidity control</td>
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<td>TMI</td>
<td>Trans-Mars injection</td>
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<tr>
<td>TMIS</td>
<td>Trans-Mars injection stage</td>
</tr>
<tr>
<td>TPS</td>
<td>Thermal protection system</td>
</tr>
<tr>
<td>TT&amp;C</td>
<td>Tracking, telemetry, and control</td>
</tr>
<tr>
<td>T/W</td>
<td>Thrust to weight ratio</td>
</tr>
<tr>
<td>UN-W/25Re</td>
<td>Uranium nitride - Tungsten/25% Rhenium reactor fuel</td>
</tr>
<tr>
<td>VAB</td>
<td>Vehicle Assembly Building</td>
</tr>
<tr>
<td>VCS</td>
<td>Vapor cooled shield</td>
</tr>
<tr>
<td>Vinf</td>
<td>Velocity at infinity</td>
</tr>
<tr>
<td>WBE₂C/B₄C</td>
<td>Tungsten beryllium cabide/Boron cabide composite</td>
</tr>
<tr>
<td>WMS</td>
<td>Waste management system</td>
</tr>
<tr>
<td>W/O</td>
<td>Without</td>
</tr>
<tr>
<td>WP-01</td>
<td>Work package 1 (of SSF)</td>
</tr>
<tr>
<td>w/sq cm</td>
<td>Watts per square centimeter (should be Wcm⁻²)</td>
</tr>
</tbody>
</table>
Z  Atomic number
zero g  An unaccelerated frame of reference, free-fall

[order: numbers followed by greek letters]

100K  \( \leq 100,000 \) particles per cubic meter larger than 0.5 micron in diameter
7n7  Where \( n=(0,2-6) \): Boeing Company jet transport model numbers
K  Kelvin (K)
+e  Positive charge equal to charge on electron
-e  Charge on electron
\( \Delta V \)  Change in velocity
s  Standard deviation
\( \mu g \)  Microgravity
I. Evolution of Concept
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Concept Development
EVOLUTION OF THE CRYOGENIC PROPULSION VEHICLES

TECHNICAL ARCHITECTURE PRESUMED LEVEL I REQUIREMENTS -

During the course of the STCAEM study, and particularly during the 90 Day Study, many SEI (then HEI) transportation requirements were generated by Office of Exploration Level II. These are reported as appropriate and necessary in various sections of this report, as well as in the STCAEM Implementation Plan & Element Description Document technical volumes. Here, space only permits a summary discussion of the Level I requirements adopted by STCAEM as they evolved during the course of the study. The concepts developed and analyzed ultimately were to accommodate the in-space transportation functions required to support the buildup of a permanent presence on the Moon and initial human exploration of Mars. Thus, our Level I requirement was simply to deliver cargo reliably to the surfaces of the Moon and Mars, and to get people to those places and back safely. Vehicles in support of missions to other destinations are not part of SEI per se, and were not addressed by STCAEM. Planet surface system characteristics and Earth-to-orbit (ETO) launch vehicle characteristics were adopted as needed for manifesting purposes, largely intact from other sources. No design work was performed for these two categories. In addition, the mission planning horizon was limited to the year 2025, about 35 years from now.

The chief Level II requirement governing the dimensions of the vehicle concepts we developed came to us during the 90 Day Study, and was a crew size of 4 for Mars missions. Subsequently, STCAEM performed a simple skill mix analysis or these long-duration missions. Our result was that doubling up on critical skills (for redundancy), given reasonable expectations of how many skills each crew member could become expert in, requires in fact a minimum of 6 - 7 crew members for Mars missions. For the sake of consistency, our vehicle concepts are shown comparable to the 90 Day Study results, sized for four crew. Impacts accruing from larger crew sizes are discussed in the Major Trades IP&ED book.

CONCEPT DEVELOPMENT METHODOLOGY - A vehicle concept emerges gradually through the iterative combination of requirements analysis, subsystems analysis, mass synthesis, performance analysis and configuration design. Because of the cascading, cause-and-effect nature of specific technical decisions in this cyclic process, the ability for a particular concept to remain fully parametric is incrementally lost, sacrificed for depth of detailing. The need to penetrate deeply even at the conceptual stage is twofold: (1) to uncover subtle integration interactions whose ramifications fundamentally revise the concept as they reflect back up the information.
hierarchy; and (2) to enable the production of graphical images of the concepts capable of being communicated widely but grounded firmly in engineering detail. If circumstances allow the concept development process to engage many cycles of reflexive adjustment, from requirements all the way down through subsystem detailing, the design oscillations subside eventually and the product that emerges is a robust and defensible concept. Basic differences in problems posed and solutions engineered lead concept developments in different directions. "Like" problems and solutions gravitate together; their recombination and resolution results in distinct, identifiable vehicle concepts which constitute vehicle archetypes. A concept is archetypal if it spawns concept progeny whose ancestry is clear, and if in so doing its salient features recognizably survive subsequent refinement, development and scaling. The ultimate purpose of the STCAEM Concepts and Evolution tasks was to generate, analyze, evaluate and describe such vehicle archetypes, and the role they could play in human space exploration missions.

The STCAEM architecture analysis identified seven major classes of transportation architecture for SEI lunar and Mars missions. Some are derived from different propulsion technology candidates; some are derived from distinct mission philosophies independent of propulsion method; most have many sub-options. Vehicle archetypes are keyed more closely to propulsion method than to mission mode, however, so we found that all seven SEI transportation architectures can be accomplished by derivative combinations of just five archetypal Mars transfer vehicle (MTV) concepts, two archetypal Mars excursion vehicle (MEV) concepts, and one archetypal lunar transportation family (LTF) concept. The concept evolution of these archetypes is outlined in Section x.2.

DESIGN AND NECKDOWN CRITERIA - STCAEM concept development was punctuated by four "neckdowns", which winnowed down the option candidates generated at each successive level of detail throughout the study. The four neckdowns were intended to result in: (1) feasible options, based on promising propulsion technologies capable of performing SEI-class missions; (2) preferred options, representing the handful of candidates whose performance and technological readiness were judged to warrant detailed study; (3) integrated concepts, vehicle archetypes developed sufficiently to uncover their major integration concerns and architectural context; and (4) detailed concepts, based on the reconciled integration of traded subsystems. The 90 Day Study occurred such that the first two neckdowns were effectively reversed; cryogenically propelled, aerobraking technology was necessarily preferred at that time, due to depth of understanding. However, STCAEM later rounded out the picture by completing all four neckdown activities, in an ongoing manner throughout the study.
Studying the program architecture implications of various technology options for SEI missions led to the conclusion that the most generally accessible discriminators, cost and risk, are driven by more subtle technical discriminators than, for instance, initial mass in low Earth orbit (IMLEO). These can be grouped into three broad categories: feasibility, flexibility, and multi-use design. As indicated above, feasibility was the first filter for all concepts considered by STCAEM. Flexibility has three components: (1) robustness, which is the ability to perform nominally despite variable or unanticipated conditions; (2) resiliency, which is the ability to recover from accidental delays or mishaps; and (3) evolution, which is an adaptation over time to changing requirements. Flexibility is thus a measure of a program's technical strength and safety in the face of variable extrinsic factors. Multi-use design has two components: (1) re-usability, which means using the same hardware item more than once; and (2) commonality, which means using the same hardware design in more than one setting. Multi-use design is thus a measure of a program's cost-effectiveness and intrinsic longevity. These two key architecture drivers were paramount in interpreting the results of STCAEM's technical trade studies, and figured prominently in the development of element concepts.

MARS TRANSPORTATION - Four Mars transfer propulsion candidates survived all STCAEM neckdowns: cryogenic chemical, nuclear thermal, nuclear electric, and solar electric. Analysis of aerobraking resulted in two performance ranges of interest for Mars entry (hypersonic \( \text{L/D} = 0.5 \) and \( \text{L/D} = 1.0 \)), as well as the use of high-energy aerobraking (HEAB) for capture at Mars. Consequently, the five archetypal MTV concepts are based respectively on: cryogenic/aerobraking (CAB), cryogenic all-propulsive (CAP), nuclear thermal rocket (NTR), nuclear electric (NEP), and solar electric (SEP) propulsion technologies. The two archetypal MEV concepts are based on the "low" and "high" L/D performance ranges analyzed.

Cryogenic/Aerobraked Mars Transfer Vehicle (CAB) - NASA selected cryogenic chemical propulsion, augmented by aerobraking for capture and landing at Mars, as the opposition-profile baseline for the 90 Day Study. The archetype which first resolved the dominant configuration complications for CAB Mars missions already existed (Boeing, 89). With this foundation the 90 Day Study was able to progress rapidly into performance, subsystem, operations and programmatics analyses. The 90 Day Study exercise in turn enabled refinement and validation of the CAB archetype. The major drivers for the CAB archetype are:
1) High-thrust chemical propulsion: engine-out design to accommodate shifting vehicle mass center as the mission progresses, given the fact of engine clustering and limited gimbal angle; propulsion system geometry for in-flight testing before critical mission maneuvers; and avoidance when possible of aerobrake penetrations.

2) High-energy aerobraking: current understanding of aftbody wake closure geometry, and aerodynamic simulation-based constraints on mass center location; mutual independence of MEV and MTV during final approach to Mars space, since each is captured separately; packaging of the entire MTV system in as small a capture aerobrake as possible; potential requirement for MTV brake retention and re-use for Earth capture upon return.

3) Rotating artificial gravity: physiological constrains drive the CAB archetype toward deployable tether schemes because of the effort to make the aerocaptured vehicles as compact as possible. This makes the physical arrangement of the MTV systems difficult, given both a requirement to maintain all habitable volumes (both the MTV habitat and the MEV crew cab and surface module) contiguous during transfer, and the fact that the only rotation countermass available on the return leg is the empty MTV TEI propulsion system.

4) Modular vehicle design, in an effort to maximize system commonality, to standardize integration and operations protocols, and especially to accommodate the widely varying energy (propulsive) requirements of opposition-class missions. In STCAEM, opposition missions were designed to collect most of the energy difference in the TMI $\Delta V$. This burden was more easily accommodated by the TMIS, which became a highly modular vehicle system.

5) Robotic-mediated operations: facilitating machine access into the densely packaged systems of the CAB vehicle, and designing provision for robotic EVA maintenance during the mission, is a tough but essential requirement. We baselined an operations concept in which manipulator systems could travel around the rims of the rigid aerobrake structures, both to assist in assembling the vehicles at Earth and to service them en route.

A concept called the "Shuttle-Z 3rd Stage" was detailed in response to a Level II trade. This is a modular version of the TMIS, in which each section uses its engine twice (once for ETO orbit insertion and again for the Mars departure burn). The fundamental problem with the scheme is that, with engines located on each TMIS section instead of clustered in the center, mass-balanced
engine-out on TMI is not possible without the addition of an extremely long (120 m) truss to separate the TMIS from the payload mission vehicles.

A configuration trade analysis revealed that avoiding the need for Mars orbit rendezvous upon arrival between a separate MEV and MTV by configuring one large, aerocaptured vehicle was not practicable (either a very large aerobrake, or a reconfigurable cryogenic propulsion system, appeared necessary).

An Earth-Mars cycler vehicle capable of providing periodic transfers between the two planets is one potential mission mode addressed by our architecture assessment. Such a vehicle could take a variety of forms, but for SEI-class missions, the basic function could be accomplished with a variation on the CAB vehicle. For the conventional cycler profile, aerocapture energies for the "taxi" craft needing to get into parking orbits at Mars are quite high. Re-usable vehicles for this job would probably require heavy and/or complex thermal protection systems.

OTHER SYSTEMS:
Cryogenic All-Propulsive Mars Transfer Vehicle (CAP) - The CAP archetype is fundamentally a variation of the CAB archetype, but is reported here as a separate archetype because its mission philosophy is quite distinct. The CAP concept was developed in response to two drivers:

1) Exploration of alternative purposes for SEI Mars missions led, after the 90 Day Study, to more in-depth discussions of the merits of conjunction vs. opposition profiles. Initial presumptions favored short total mission durations; this approach remained typical after the FY88 and FY89 OEXP study cycles, in which very short, compressed opposition or "split-sprint" mission modes figured prominently. However, given the 30 - 60 d Mars staytime realistically permitted by their astrodynamics, the ratio of usable surface time to total mission time for opposition profiles is about 10%. After the 90 Day Study, this was recognized more widely as a relatively disappointing science return on a large engineering investment, exacerbated by the possibility of extrinsic events (like Martian dust storms) precluding landing altogether. By comparison, the same ratio for a typical conjunction mission is about 30%. The top-level costs associated with exploiting the greater opportunity to do in-depth science proffered by conjunction missions are two: (1) the requirement for more elaborate surface payload manifests to support both that science and the crews to conduct it for year-long stays; and (2) the greater risk to mission completion incurred by having the crews and hardware spending almost 3 yr in deep space instead of about 1.5 yr.
The conjunction profile offers other benefits recognized much later. First, the opportunity variation in mission energy requirements is much reduced for the conjunction case, so that mission hardware can be more consistent from one opportunity to the next. This would minimize the actual program upset resulting from a missed opportunity. Second, having of order 300 d available at Mars would permit more flexible mission design. For example, rather than spending the entire staytime on the surface, the mission might carry multiple landers each destined for short visits to widely separated surface sites (or crew rescue at a given site). And finally, although conjunction missions are roughly twice the length of opposition missions, the bulk of that difference can consist of time spent on the surface of Mars, under the radiation shielding afforded by the martian atmosphere. The actual in-space transfers are about equal in length outbound and inbound, and their total is less than the total in-space transit time for typical opposition missions. Thus in scenarios required to minimize astronaut exposure to in-space galactic cosmic radiation (GCR), well-designed conjunction missions are of great interest. (Trip times can be shortened further still, until the so-called "conjunction fast transfer" mission energy requirements approach those for opposition missions.)

Conjunction low-energy missions do not benefit from HEAB, so these missions need only carry aerobrakes for entry and landing. Performing Mars capture with cryogenic chemical propulsion leads to three fundamental distinctions between CAP and CAB concepts:

1) The MTV and MEV(s) are captured together, precluding the possibility of failure to rendezvous and consequent scrub of landing attempts.

2) The Earth-departure (TMI) stage grows into a multi-staged propulsion stack, with TMIS, deep-space burn (DSB) stage, and Mars arrival (MOC) stage. This changes the overall aspect ratio of the all-up vehicle, making it longer, which has implications for attitude control and debris shielding in LEO.

3) Relaxing the requirement for the MTV to be an aerobraked vehicle means that the systems constrained in the CAB case to be packaged behind an aerobrake can be distributed differently. Thus the Mars-departure (TEI) propulsion system can be combined with the MOC system and placed at the opposite end of the vehicle from the MTV habitation system and payload. This in turn means that rotating artificial gravity can be accomplished as simply as for the NTR vehicle, by configuring a long, lightweight truss between the propulsion end and the payload end, and spinning this rigid assembly end-over-end. Tethered solutions are not required because aerobrake
packaging is no longer a problem. This last set of CAP consequences departs from the CAB concept sufficiently for their resolution to constitute a distinct vehicle archetype.

ARTIFICIAL GRAVITY (CAB) - The need for artificial gravity on long-duration interplanetary transfers has not been established. Neither has the lack of such a need, however, so STCAEM was obligated to examine the penalties incurred by requiring continuous artificial gravity en route between Earth and Mars. Various approaches to rotating artificial gravity have been proposed; STCAEM assessed all of them, and invented some new ones. The fundamental design problems associated with artificial gravity derive from: (1) the need for a countermass for rotation; and (2) the high mass cost of precessing the angular momentum vector of a system having large rotational energy. Elegant solutions to both are elusive, and vary widely with propulsion option. Secondary complications are communications and navigation pointing, flight structures sized to hang heavy vehicles, and possibly material fatigue. The fundamental operations problems associated with artificial gravity involve crew EVAs during rotation, robotic maintenance in the vehicle's gravity field, crew physiological and psychological responses to a rotating environment, performing minor course-correction propulsive maneuvers and testing the capability prior to departure. Our work has verified that artificial gravity appears feasible for Mars-class missions, for all propulsion options, at fairly modest mass penalties.

The CAB archetype involves more complexity. The MTV habitat must be contiguous with the MEV crew modules, and yet for the return trip the (empty) MTV propulsion system is the only available countermass to the MTV habitat. Thus the MTV hab and the MTV propulsion system must be separated by a few hundred meters; however, the entire MTV must also package behind an aerobrake for capture at Mars. One solution we rejected for mass and habitability reasons splits the transfer habitat system in two halves, held when not aerobraking at opposite ends of a deployable tunnel. A more sensible approach is to use tethers, configuring the MTV systems such that they are properly mass-balanced for propulsive burns and aerocapture, but can slip apart as the tethers are unreeled for artificial gravity. The center of rotation provides a convenient location for a despun power/navigation/communications utility.

ARTIFICIAL GRAVITY (CAP) - .The CAP and NTR archetypes accommodate artificial gravity easily. Both are high-thrust systems, so their burn times are extremely short (minutes to hours) compared to coasting transfer time (months). Critical propulsion maneuvers can occur during nonrotating periods of microgravity, at the cost only of spinup/spindown propellant. In general, the propulsion system remaining through the end of the mission can serve as countermass
to the contiguously connected habitation systems. When separated by a lightweight truss, they can just spin end-over-end during coast phases to provide sufficient gravity at a comfortable spin rate with acceptable vestibular disturbance (we baselined 1 g to insure full conditioning for surface activity upon arrival at Mars, and 4 rpm maximum spin rate, which together lead to a 56 m separation between the hab and the center of mass). The additional mass of the truss and propellant for a few budgeted spinup/spindown cycles is of order 10 % of IMLEO.
Low-L/D Mars Excursion Vehicle (MEV) - The MEV archetype development began during, and was resolved just following, the NASA 90 Day Study. It was originally conceived as a means of delivering 25 t of undefined payload to the surface of Mars. However, the specification of crew cab provisions, the analysis of vehicle mass balance, and consequently the configuration design of the vehicle all depend on specifics of the payload manifest. We assumed a 20 t reference surface module as an integral part of the MEV. This led to a "Mars campsite" design intended to support a crew of four for 30 - 60 d and became our standard lander design. Chief departures from the lunar campsite mode of operation were:

1) The MEV arrives with the crew already onboard, and so is capable of a really self-contained mission.

2) The MEV also brings with it an ascent vehicle (MAV) with a separate propulsion system, configured optimally for the ascent phase (or ascent after breakaway from the descent stage during a descent abort). The crew cab for the MAV is the operations bridge for the MEV during all its mission phases.

3) The MEV is configured for packaging within an L/D = 0.5 aerobrake. For CAB missions, this brake captures the as-yet unmanned MEV into Mars orbit autonomously, before rendezvous with the MTV, and is used again for the descent. For CAP and other types of missions with propulsive Mars orbit capture, this brake is used only for descent. In all design cases, terminal descent engines are extended through ports in the windward surface of the brake at low Mach number, and the brake is jettisoned subsequently, prior to touchdown.

The MEV configuration was developed to permit later removal and relocation of the surface habitat module, with the aid of surface construction equipment. A variant of the MEV, without either surface module or MAV, was analyzed for delivery of heavy cargo on unmanned missions. A quick assessment was made of the feasibility of re-using an MEV, presuming in situ production of oxygen and retention of the aerobrake until touchdown. The outcome was positive, although: (1) additional brake hatches appeared necessary for landing gear deployment, crew egress, and cargo offloading; and (2) a lightweight top-shroud appeared advisable due to aerodynamic drag on ascent, and to permit the crew bridge to protrude beyond the presumed wake-protection limit for direct surface viewing during terminal approach. Configuration options for a "split-stage" MEV,
in which the same, or a portion of the same, propulsion system is used for ascent as for terminal descent, were also investigated, and shown to be simple variations of the archetype.

Our baseline aerobrake assembly concept presumed robotic-mediated final assembly of pre-finished, rigid aerobrake segments at Freedom. Packaging such segments efficiently by nesting them in an ETO launch shroud is made challenging because of: (1) the aerobrake's asymmetrical, deep-bowl shape, in which the maximum depth of a typical "slice" is comparable to reasonable shroud diameters; and (2) the aerobrake's lip, required for both aerodynamic performance and structural stiffening around the free brake edge. Subsequent manifesting analysis, in which segments were configured according to an initial rib-and-spar structure concept, indicated that two ETO flights would be required to launch a single aerobrake in several pieces. Such extremely volume-limited and volume-inefficient manifesting is an unacceptably poor use of the expensively developed capability that a heavy-lift ETO system represents.

In response to this manifesting problem, STCAEM proposed the "integral launch" concept, in which a fully assembled, integrated aerobrake is launched externally, mounted on the side of the launch vehicle exactly analogous to current STS operations. The low-L/D brake is comparable to the STS orbiter in linear dimensions, and is light enough to launch two at once, with capacity to spare for other, shrouded payload as well. Ascent performance of such a flight configuration requires study; the critical question is whether ascent loads would size the aerobrake structure out of the competitive mass range for the mission itself.

Our structural analysis indicates that since the deep bowl-shaped aerobrake loads like a doubly-curved shell, it may be possible to construct an actual "aeroshell" without resorting to ribs and spars or some other articulated skeletal structure system. The shell would be made of a relatively thin honeycomb-type material system with integral TPS. However, lip buckling would still require a stiff rim, probably facilitated by a closed-tube-section structure. Such a brake may be lighter, and certainly simpler, but the thickened rim would still cause packaging problems due to nesting interference.
High-L/D Reusable Mars Excursion Vehicle (RMEV) - The RMEV archetype development occurred in response to three drivers:

1) Analysis so far indicates that L/D = 0.5 is sufficient at Mars for controlling an aero-vehicle at Mars. However, the existence of some mission design studies in the literature which advocate L/D > 1.5 for Mars, combined with our preliminary understanding of controllability under Mars conditions, make it important to know in detail how different the configuration constraints imposed by higher L/D would be from those imposed by the lower L/D (which by 1989 had come to be regarded generally as appropriate).

2) As the 90 Day Study stimulated thinking about what the purpose of SEI Mars surface missions should be, concern developed that global, or at least wide, access to the surface of Mars was potentially important. High-thrust Mars transfer propulsion systems (chemical or NTR) tend to be mass-constrained by arrival and departure vector geometry to certain parking orbit conditions. Although there is no lack of interesting (scientifically important) landing sites accessible from the periapsis of any orbit at Mars, the fact that performance-optimized parking orbits are unique for each high-thrust opportunity causes a site-access problem if returning to the same surface site is required (for base buildup). Thus for high-thrust transfer propulsion options particularly, an ability to achieve cross-range on lander entry may be important. High L/D enables greater cross-range capability.

3) Certain Mars lander issues not imposed as requirements during the 90 Day Study required analysis and design validation. Developing a new MEV concept, substantially different from the baseline MEV, allowed us to investigate those issues simultaneously and thoroughly. Specifically, we addressed: (1) a deep aerobrake structure concept, of interest for maximum structural efficiency and therefore reduced brake mass; (2) the ability to deliver large-envelope cargo manifests, represented in our design by a long-duration surface habitat module sized for 10 crew; and (3) re-usability of the MEV, based on in situ production of cryogenic propellant.

The vehicle shape represented by the RMEV has applications for other interesting mission modes, concepts for which have yet to be investigated in detail. Three examples are: (1) a smaller RMEV, sized commensurately with the MEV to be a modest cargo-delivery vehicle; (2) a direct-landing MTV, whose return propellant would be manufactured in situ on Mars; and (3) re-usable aerobraked "taxi" vehicles capable of performing the Earth-Mars cycler embark/debark function.
Cryogenic/Aerobrake (CAB) Reference Configuration

Introduction

The cryogenic/aerobrake (CAB) concept was used as the NASA 90-day Study reference vehicle. It offers conceptual continuity with the mainstream Mars transportation studies performed over the last several years. Its only major new technology development is high energy aerobraking (HEAB) for planetary capture, but the concept also requires a high-thrust cryogenic space engine. Being able to land on Mars using the CAB concept requires a successful rendezvous between separately captured vehicles in Mars orbit.

Nominal Mission Outline

- The vehicle is assembled, checked out and boarded in LEO
- The TMI burn occurs and the TMIS is jettisoned
- MTV/MEV coast to Mars
- MTV and MEV separate 50 days prior to Mars capture
- The MEV aerocaptures robotically a day ahead of the MTV, providing last-minute verification of atmospheric conditions and targeting
- The MTV captures, followed by rendezvous in the parking orbit with the MEV
- The landing crew transfers to the MEV and checks it out
- The MEV descends to the surface, jettisoning its aerobrake prior to landing
- After surface operations, the ascent vehicle (MAV) leaves its descent stage and surface payloads, ascends to orbit and docks with the MTV for crew transfer
- The MAV is jettisoned in Mars orbit, and the TEI burn occurs
- The MTV coasts back to Earth
- The crew transfers to a modified ACRV (MCRV), jettisons the MTV and performs a direct entry at Earth (optional: the entire MTV aerocaptures into a LEO parking orbit for refurbishment and re-use)
Vehicle Systems

The vehicle consists of three main elements: the Mars Excursion vehicle (MEV), the Mars Transfer Vehicle (MTV) and the Trans-Mars Injection Stage (TMIS).

Mars Transfer Vehicle (MTV)

The MTV configuration shown consists of a transit habitat sized for 4 crew, an aerobrake, and a TEI propulsion system. The transit hab is located centrally in the aerobrake with an external airlock and an MCRV attached to the top (in the configurations shown, an Apollo-style ECCV was used to represent the MCRV). The airlock allows access to the MEV crew cab and surface habitat during all phases of the transfer mission until the MEV separation 50 days prior to Mars arrival. The MCRV is used for mission scenarios featuring direct-entry crew return; these scenarios expend the entire MTV upon return to Earth. In a reusable mode, the entire MTV would be aerocaptured back at Earth for refurbishment and re-use; a second airlock would be located in place of the MCRV. The aerobrake is of identical geometry and construction as the MEV aerobrake, but is stronger and heavier due to its larger payload mass, and does not require any engine doors. The propulsion system (TEI) is divided symmetrically into two tank-stacks straddling the transit hab, like the MAV tankset configuration. The propulsion system is oriented at an angle relative to the aerobrake axis, with the two engines aimed out the rear of the aerobrake, to avoid TPS penetrations while still permitting mass-balanced operation during the burn.

Trans-Mars Injection Stage (TMIS)

The TMIS consists of a core unit with four advanced space engines (ASE), avionics and cryogenic propellant tanks, and provision for up to four “strap-on” propellant tanksets. This configuration allows propellant cross-feeding in the case of engine-out, and modular accommodation of the entire stage’s performance according to the mission opportunity requirements. Keeping the engines close
together on the core stage allows tracking the CM during an engine-out condition via gimballing. This strategy avoids either opposite-shutoff (leading to long burn times and greater gravity losses), or a requirement for extra structure (a 125m truss) between the propellant tanks and engines to allow CM tracking. The TMIS accounts for about 75% of the total IMLEO, a substantial per-mission resupply cost.

**Mars Excursion Vehicle (MEV)**

The reference MEV is a manned lander that can transport a crew of 4 to the surface. It consists of a surface-stay habitat module (roughly SSF-module size), an airlock, 5 t of surface-science payload, a cryogenic descent propulsion system with four engines and bus structure, and the ascent vehicle (MAV). The MAV consists of a short-duration crew cab, and cryogenic ascent propulsion system with two engines. All propellant tanks are mass-balanced around their maneuver CMs so that no lateral CM shifting occurs. The entire MEV is packaged in a rigid, truncated-hyperboloidal aerobrake with \( L/D = 0.5 \), to which it is attached at eight points (four bus-frame corners and four landing-gear footpads). The aerobrake is fitted with doors which open to allow the descent engines to extend and ignite prior to aerobrake separation (allowing full benefit of the brake’s drag). The brake is then jettisoned as the landing gear extend prior to terminal approach and hovering touchdown.

Dominant configuration constraints for the MEV are as follows:

- Payload manifesting
- Surface access
- Crew visibility
- Contiguous crew volumes
- Short vehicle stack
- Engine-out capabilities
- On-orbit assembly

Payload manifesting is mainly a proximity and mass balance issue. The surface habitat and airlock, which is the bulk (80%) of the payload, require access to the ascent crew cab and the surface, as well as being mass balanced for proper flight. The science payload requires surface access for ease of unloading. Docking is
facilitated by placing the crew cab high in the vehicle stack. The flight deck window is located to provide viewing to the surface for landing as well as to the upper hatch for docking. Keeping crew volumes contiguous allows access during flight for check-out procedures and simulation training. The vehicle stack is kept as short as possible for aerobrake wake protection, which tends to conflict with having the center of mass (CM) as high as possible, desirable for a small engine gimbal-angle to provide minimal steering loss in an engine-out scenario. A high CM within a short stack is accomplished by placing the dense ascent LOX high in the configuration. Finally, although the dominant constraints for the MEV derive from its performance at Mars, consideration has been given to its ETO launch. It is configured to be launched in a few, large, pre-integrated systems for minimal on-orbit assembly. For example, the ascent vehicle can be launched intact in a 10 m diameter shroud, while the descent structure can be launched in 2 sections for fairly simple on-orbit assembly and integration.
Mars Mission Vehicle in LEO

The Mars vehicle LEO configuration is shown here ready for trans-Mars insertion (TMI).

The TMI stage launches the vehicle out of Earth orbit on a trans-Mars trajectory. There are four propellant tanks and five engines in the TMI stage; it is modularized for compatibility with the launch vehicle. The elements of the TMI stage are launched fully loaded with propellant.

The Mars excursion vehicle includes an aerobrake for Mars capture and entry/landing, a descent propulsion stage, an ascent propulsion stage with crew module for Mars descent, ascent, and contingency surface operations, and 25 t. of surface payload (a habitat and science) for normal surface operations.

The Mars transfer vehicle includes its own aerobrake for Mars capture, a long-duration crew habitat for the trips to and from Mars, a propulsion system for boost out of Mars orbit to return to Earth, and the Earth crew capture vehicle. The TMI stage is bookkept as part of the Mars transfer vehicle for WBS purposes. On some missions, the MTV aerobrake returns to Earth with the vehicle so that the MTV (except for the TMIS) can be captured in Earth orbit for reuse on another mission.

All crew volumes are contiguous between the MEV and MTV during TMI and coast.

The mass totals for option 1 and 5 are shown for comparison. The only difference between options 1 and 5 is that option 5 carries a surface reconnaissance vehicle into Mars orbit on the MEV (it is not shown on the chart). The surface reconnaissance vehicle is launched from the Mars parking orbit to perform robotic exploration of a future human landing site.
Cryo/Aerobraking

ADVANCED CIVIL SPACE SYSTEMS

Trades and Rationale

- A core stage with four advanced engines and four "plug-in" propellant tanks. Tanks and core stage rendezvous and dock automatically. Core stage provides simple plumbing and good engine out performance.

Mission Modes And Operations

- NASA 90 day study baseline.
- Vehicle assembled in SSF orbit.
- TMIS abandoned after TMI burn.
- MEV/MTV separate prior to Mars aerocapture.
- Crew transfer to MEV/Aerobrake after MTV/MEV rendezvous.
- MEV/Aerobrake entry. Aerobrake jettisoned prior to landing.
- Crew cab ascent after surface mission, leaving lander and surface hab.
- Crew cab left in Mars orbit after rendezvous, docking and crew transfer.
- TEI burn.
- Crew return to SSF after aerocapture.
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Cryo/AB Reference Configuration

ADVANCED CIVIL SPACE SYSTEMS

MTV total 163.7 t
MEV total 84.4 t
3CCV 7.0 t
Interstage Structure 0.5 t
FM1 stage total 545.5 t
MLEO 801 t
MTV Reference Configuration

The facing page illustrates the reference MTV / Aerobrake configuration. Shown are the transfer hab, ECCV, airlock and TEI propellant tanks and engines. The tank and engine structure is configured as shown to allow docking access by the MEV, and to allow the assembly to remain within the protected wake region.
Options / Alternatives
Options and Alternative Configurations

Alternative Landers

As an alternative to using the 0.5 L/D hyperboloid shaped aerobrake for a landing vehicle, investigations were made using a high (1.0+) L/D lifting body aerobrake shape and a Bi-conic shape. Both of these shapes extend the crossrange capability and are candidates for a reusable Mars Excursion Vehicle (RMEV), the criteria for which is given on the next pages. It appears that the high L/D aerobrake will be better suited for a reusable system, with fewer specialized parts. The Bi-conic will impose some restrictions on cargo that high L/D aerobrake will not, such as the delivery of a 10 crew habitat to the surface. In the case of the Bi-conic the habitat would have to be either specially built to fit the available space or the entire fleet of habitats would be scared to have this shape, at additional cost of fabrication. Other constraints became evident, while the high L/D aerobrake has limited visibility of the ground during landing operations, the Bi-conic has none.

Alternate Mission Vehicles

An all-propulsive cryogenic (chemical) vehicle was evaluated for near term conjunction missions. Conjunction missions were chosen due to the significant increase in IMLEO to mount an opposition mission using an all-propulsive vehicle (~ 1600 t IMLEO), while a conjunction mission would be in the range of the cryo/aerobrake mission under consideration (625-720t IMLEO). The advantage of the all-propulsive vehicle is that it has a short development time and can be ready early. The disadvantages are the limitation to the conjunction-class missions (long stay times at MARS, total trip time is long) and only the ECCV is recoverable, all other sections of the vehicle are expended in operations.
Reusable MEV Status

**Reusable MEV Parametrics**

**Assumptions**
- Landing propellant and ascent H2 supplied in Mars orbit
- Ascent O2 from mars surface.
- 470 Isp
- Down payload 29.5 t. (25 t. + 4.5 t. c/m)
- Ascent payload 4.5 t. crew module
- Descent: aerobrake + 1200 m/sec
- Ascent ΔV 5500 m/sec
- Mixture ratio 6:1

**Configuration Issues**
- Location of payload at c.g. for landing
- Payload removal method
- Location of crew module for landing visibility
- Location of engines for landing and ascent
- Location and arrangement of tanks
- Landing gear design and placement for landing and ascent
Reusable MEV Sensitivities

**Advanced Civil Space Systems**

**BOEING**

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**Direct Return To Earth**

**Direct Payloads**
- 40 t. landed
- 40 t. return, except as noted

**Mars orbit payloads**
- 30 t. landed
- 5 t. return

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**Expendables**

- Ref MBV (expendable)
- Mini Mev (expendable)
- Direct Ref (Mars propellant)
- Direct
- 25 t. Cargo (65 t. landed)
- 20 t. return payload
- 10% Brake
- Methane 5% tank
- Meth 20 t. Ret.
- Meth 4% tank
- Mars Orbit 1-sol
- Mars Orbit 10-hr
- Mars Orbit 10-hr 7:1 mixture ratio

**Reusable**

- 15% Aerobrake direct
- 10% Aerobrake from Mars orbit
- 8% Tank fraction for O2-112

---

D615-10026-2

/STCABM/prw/15June90
High L/D Aerobraking Constraints

Shown on the facing page are constraints as applied to the high L/D aerobrake, which were used to configure the reusable MEV. Aerobraking constraints include resultant force vector and protected wake cone, which impact the location of the MEV within the aerobrake.
High L/D (1.1) Aerobraking Constraints

ADVANCED CIVIL SPACE SYSTEMS BOEING

Geometric Axis

Protected Wake Cone

Free Streamline

50° RWA

Resultant Force Vector

32 m
High L/D Reusable MEV Configuration

The high L/D reusable MEV is shown on the facing page. This configuration allows offloading of payload (shown here as a 4.4 m dia. hab module) by way of a ramp and track system, located at the rear of the vehicle. The ramp also acts as a body flap for aero-maneuvering. The crew cab is positioned so that the pilot has down and forward visibility until the landing area is selected. As the vehicle rotates into landing attitude, crew visibility will be limited to the surface directly below.
High L/D (1.1) MEV Configuration

- Payload at CM and removed out back of aerobrake via track system. Back landing legs retract to lower body flap to surface for payload removal
- Crew visibility for landing accommodated through front landing leg door
- Vehicle mass balanced to allow for flight with or without payload
The Bi-conic lander/habitat is configured to be launched atop a 12m dia. HLLV, and delivered to Mars orbit via a separately launched TMI stage. The Bi-conic shape provides an L/D of 1.3 at a 20 degree angle of attack, and lands using a 6 engine configuration, split 3 forward and 3 aft. The unmanned vehicle is used in an expendable mode, and requires 21 metric tons of propellant for landing.

The 10 crew habitat module delivered to the surface is integrated within the bi-conic, and would not need heavy transportation equipment for deployment. The hab module weighs 40 metric tons when landed, and would need to be outfitted on the surface.
Bi-Conic Lander / Habitat

- Science/payload
- 10.5 t propellant
- Hab area, 100m² per crew (10)
- LH₂
- LO₂
- Wardroom/galley
- Crew quarters
- Science/work
- Science/EVA
- BCLSS
- Resultant force vector for 20° angle of attack
All-Propulsive Cryogenic Vehicle

**Trades and Rationale**

- Addition of MOI/TEI stage eliminates the need for a high energy aerocapture at MARS
- ECCV return for crew eliminates the need for a high energy aerocapture at Earth
- ECCV return is direct entry (Apollo-style, done before)

**Mission Modes and Operations**

- Vehicle assembled in SSF orbit.
- TMIS discarded after TMI burn.
- MOI burn and capture prior to MEV/aerobrake entry
- Aerobrake separates from MEV prior to landing.
- Crew cab ascent after surface mission, leaving lander, surface habitat
- Crew cab left in Mars orbit after rendezvous, docking and crew transfer.
- TEI burn.
- Crew return to Earth via ECCV
An Propulsive Chemical Vehicle for Conj Mission

Single MEV; 5t surf cargo, crew of 4, Common tank sets for MTV stgs
dV's; TMI dV= 3900 m/s, MOC=1530 m/s, TEI= 860, E arr Vinf=3200, TMI, MOC, TEI eng Isp=475, MEV eng Isp=460

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<thead>
<tr>
<th>Element</th>
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<td>[378] MTV crew hab module 'dry'</td>
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<td>[398+371] MTV hab consumables &amp; resupply</td>
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[171] IMLEO (all masses in kg) | 624117

5/7/90
### Propulsive Chemical Vehicle for 2010 Conjunction Mission

Single 73t MEV carries 25t to surf, Common tank set for MOC/TEI stg

dV's: TMI dV= 4570 m/s, MOC=1160, TEI= 1186, Adv space eng's: Isp=475

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**Diagram:**
- Crew return via ECCV, no vehicle reuse
Single MEV ;30t surf cargo, crew of 3 for 90 days on surf, 1 crew member left in orbit, Common tank sets for MTV stgs dV's: TMI dV= 3900 m/s, MOC=1530 m/s, TEI= 860, E arr Vinf=3200,TMI, MOC,TEI eng Isp=475, MEV eng Isp=460

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<td>[63]</td>
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<td>[171]</td>
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# Cargo Chem/aerobrake Veh for one way 2018 Conjunction Mission

Unmanned, 2 cargo landers (46.5 t surf cargo each), 10 t navigation set, no MTV propulsion stg, TMI stg Isp=475

**Revision 2 7/30/90**

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Mac chart: Cargo chem/ab 2018 wt cover pg
Veh synthesis model run #: marschemmtv dat:37
# Common Mars/Lunar Lander Vehicle - Cargo & Manned Versions

*Mars desc propul dV: 773, Asc dV: 5319, Lunar asc & desc dV: 2100, all cryo prop Isp=475*

Single stage vehicle - aeroshell, cargo and landing legs left on surface

<table>
<thead>
<tr>
<th>Element</th>
<th>Mars Cargo (desc only)</th>
<th>Mars *Manned (single stg desc/asc veh)</th>
<th>Lunar Cargo (desc only)</th>
<th>Lunar *Manned (single stg)</th>
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* Manned: crew of 3 or 4 for very short surf stay time (a week or less)

** Maximum surface cargo load for manned lunar case when all tanks are full
Architecture Matrix
Reference Matrix to Alternative Architectures

In considering a complex task, it is useful to organize it into a hierarchy of levels. The higher levels are more important or more encompassing, while the lower levels include more detail or are more specific. Constraints (e.g., requirements and schedules) flow down from the higher levels and solutions or implementations build up from the lower levels. The first figure shows a hierarchy of six levels from national goals to performing subsystems. The following section discusses the fourth level, exploration architectures, in terms of the lower levels: element concepts and performing subsystems. Selection of preferred architectures will require the Government (the National Space Council, the President, and the Congress) to first define the top three levels.

Implementation Architectures

Seven architectures have been selected for examination: four different propulsion types (Cryogenic/Aerobrake, NEP, SEP, and NTR); two variations of In-Situ Resource Utilization (ISRU) for propellants with Cryogenic/Aerobrake propulsion (Lagrange point 2 refueling and Mars surface refueling); and a cycling spacecraft concept. Three basic levels of program scope are identified: small, moderate, and ambitious.

Multiple options can be generated within the basic architectures, varying launch vehicle capacity, orbital node type, and mission profile and propulsion type for the various Lunar and Mars vehicles.

Aerobraking is found to be applicable to all seven architectures, placing it as a 'critical' technology. Electric propulsion leads to the lowest reference vehicle mass, and also almost the lowest resupply mass. ISRU/Cryo leads to the lowest estimated resupply mass since most of the propellant is derived locally rather than coming from Earth.

Cost Models

Cost estimation is being performed using "parametric" methods. This technique uses a parameter, usually weight, as an input to empirically derived equations that relate the parameter to cost. It should be recognized that the source data for the cost models is past program experience, while the hardware being estimated will be built one or two decades from now. Therefore these cost estimates should be assumed to have a standard deviation on the order of +-100%. Hardware at technology readiness level 5 may be assumed to have a standard deviation in cost estimate of +-30%. No revenues from sale of products, services, or rights (i.e., patent rights, data rights), or commercial investment, are assumed in the cost estimates. These might appear in a scenario such as the Energy Enterprise.

As an example, the cost estimate for a NEP architecture shows an average annual funding level of $8 billion per year after initial ramp-up.

The principal cost drivers identified include number of development projects, reuseability, mass in Earth orbit, and mission/operational flexibility.

Analysis Methods

Individual trade studies are performed within each architecture to optimize it against evaluation criteria. The principal evaluation criteria to date has been initial mass in low Earth orbit, as a proxy for cost. The results of this optimization will then be compared to each other in groups. The early Mars group will compare all-propulsive, aerobraking,
direct travel, and nuclear thermal among themselves. The electric propulsion group will compare SEP and NEP. The innovative group will compare Lunar oxygen to cycler orbits. These concepts may both be retained if it is advantageous to do so. Finally, the choice between early Mars and Late/Evolving Mars will need to be made on the basis of cost, risk, and performance, while combining the best features from each group.
Logical Types for Space Programs

Architectural planning for a space program deals with many levels of information. A major space program like the space exploration initiative must respond directly to national goals in traceable ways. While we do not determine national goals, it is our business to understand how exploration architectures can be evaluated in terms of national goals.

National goals translate to space specific goals for specific exploration programs such as science emphasis or expanding human presence. These in turn can lead to program strategies for space-specific goals such as low risk, high technology, low cost and so forth. Finally, exploration architectures are integrated assemblages of systems, mission profiles, and operations, necessary to satisfy program goals.
"Logical Types" for a Space Program

National Goals

Space-specific Goals

Program Strategies

Exploration Architectures

Element Concepts

Performing Subsystems

Increasing Importance

Increasing Detail

Each logical type subsumes all the subordinate types
Overall Study Flow

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown on the facing page. Combinations of major technologies, such as electric propulsion and aerocapture, were quickly determined to be uneconomic in view of high development costs. Further, we found that electric propulsion systems could perform both crew and cargo Mars missions if crews are transported to and from the electric system at about lunar distance by a lunar transfer vehicle.

New systems introduced included nuclear thermal rocket (NTR) and Mars direct. NTR was introduced as an option by NASA during the "90-day study". We introduced the Mars direct profile (everything is landed on Mars; the return propulsion system is loaded with oxygen and perhaps fuel as well on Mars) in March 1989. Martin-Marietta subsequently publicized one variant of this concept.

Lunar oxygen for Mars missions was found to be uneconomic because of long payback time for the launch mass required to emplace lunar oxygen production on the Moon. Lunar oxygen has a reasonable return on investment for lunar transportation at two or more lunar trips per year.

The cycler architecture was broadened to include semi-cyclers. Late in the study we introduced an NTR-dash mode (described later in this briefing) closely related to the semi-cyclers.
Overall Study Flow

- NTR
- Mars direct
- Cryo Aerobrake backup
- Cryo All-Propulsive option
- NTR w/split-sprint option
- Baseline
- Mars direct
- Further Study
- SEP
- SEP Cycler
- Lunar L2 mode
- Cryo A/B SEP Cargo
- Lunar w/NEP cargo
- Lunar 2-stg
- Cryo A/B NEP cargo
- Uneconomic
- Lunar LOX for Mars uneconomic
- SEP Dark Horse
- CR&R
- Lunar Family
- Start expendable

Dropped by NASA before ATP
Program Implementation Architectures

We have selected seven program implementation architectures for architectural analysis. These seven architectures incorporate the advanced propulsion options of principal interest in complete evolutionary architectural scenarios for lunar and Mars exploration. The facing page lists the features of each architecture and the rationale for selection of each.

Some of the architectures include suboptions. For example, the nuclear electric propulsion and solar electric propulsion architectures include optional use of the electric propulsion system for lunar cargo delivery from LEO to lunar orbit. The L2-based cryogenic aerobraking architecture includes use of NTR and NEP vehicles for LEO to L2 cargo delivery as options, and also includes a cryogenic all-propulsive conjunction mission option.
# Program Implementation Architectures

**ADVANCED CIVIL SPACE SYSTEMS**

## Architecture

<table>
<thead>
<tr>
<th>Architecture</th>
<th>Features</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cryogenic/aerobraking</td>
<td>Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.</td>
<td>NASA 90-day study baseline</td>
</tr>
<tr>
<td>NEP</td>
<td>Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.</td>
<td>High performance of nuclear electric propulsion</td>
</tr>
<tr>
<td>SEP</td>
<td>Solar electric propulsion for Mars transfer; optionally for lunar cargo.</td>
<td>High efficiency of solar electric propulsion; find cost crossover for array costs.</td>
</tr>
<tr>
<td>NTR (nuclear rocket)</td>
<td>Nuclear rocket propulsion for Lunar and Mars transfer.</td>
<td>High Isp of nuclear rocket enables avoidance of high-energy aerocapture at Mars.</td>
</tr>
<tr>
<td>L2 Based cryogenic/aerobraking</td>
<td>L2-based operations; use of lunar oxygen.</td>
<td>L2 base gets out of LEO debris environment. Lunar oxygen reduces resupply by ~ factor 2.</td>
</tr>
<tr>
<td>Direct cryogenic/aerobraking</td>
<td>Combined MTV/MEV refuels at Mars and LEO. &quot;Fast&quot; conjunction profiles.</td>
<td>Eliminates Mars orbit operations.</td>
</tr>
<tr>
<td>Cycler orbits</td>
<td>Cycler orbit stations à la 1986 Space Commission report</td>
<td>Eliminates boosting massive Mars transfer vehicle.</td>
</tr>
</tbody>
</table>
SEI Program Scopes for Transportation Architecture Analysis

There are many space-specific goals and program strategies. We believe that transportation architectures will respond mainly to program scope. Some architectures are best suited to small program with early goals and others best suited to long range larger programs with ambitious goals. We have selected three representative scopes for small, moderate and large programs as illustrated on the facing page. These scopes permit definition of transportation requirements in terms of numbers of people and amounts of cargo transported to particular locations on particular schedules.

The second important feature of the scopes we intend to investigate is that they cover a scale factor greater than ten. A man tended science station may have few people on the Moon for short periods, or few people on Mars for short periods every other year. Permanent science bases will involve a dozen or so people. Industrial development of lunar resources on a scale of helium-3 scenarios leads to numbers of people presently estimated in the range of thousands by 2050. Beginnings of humans settlement of Mars involves numbers in the range hundreds to thousands. The 20-25 horizon for SEI is expected to permit growth in numbers of people only to dozens or so.
## SEI Program Scopes for Transportation Architecture Analysis

### Advanced Civil Space Systems - Boeing

<table>
<thead>
<tr>
<th>Descriptor</th>
<th>Small</th>
<th>Moderate</th>
<th>Ambitious</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lunar Operations</td>
<td>Man-tended science station</td>
<td>Permanent science base 6 - 12 people</td>
<td>Industrial development of lunar resources</td>
</tr>
<tr>
<td>Mars Operations</td>
<td>Expeditionary visits ~4 people</td>
<td>Permanent science base 6 - 12 people</td>
<td>Beginnings of human settlement</td>
</tr>
</tbody>
</table>
Three Activity Levels for Architecture Evaluation

We established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in-space transportation technologies as baselines for greater activity levels.

Activity levels were selected with underlying program objectives in mind:

(1) The minimum lunar program establishes astrophysical observatories on the Moon and provides a man-tending capability to maintain them. To the extent that man-tending lunar visits are not needed for the observatory system, the transportation capability can be used to explore interesting lunar sites for lunar geoscience objectives.

(2) The minimum Mars program is very similar to Apollo, i.e. six sites visited for short periods (two sites per mission and three missions); samples obtained within a few km. of each landing site. If the manned visits are preceded by suitable robotic missions, the scientific payoff for these visits can be high relative to the investment.

(3) The "full science" lunar program adds human permanence at the Moon for extensive scientific and technological exploration. Where the minimum program offers very little opportunity for lunar geoscience, this program offers more. It also permits development of in-situ resource technology for production of surface systems. The reference program also emplaced a lunar oxygen production system to serve the transportation system.

(4) The "full science" Mars program multiplies by several crew person-days on Mars by including more missions and by more staytime per mission. This program falls short of a permanently-occupied base on Mars, but achieves surface stays greater than a year.

(5) The lunar industrialization program adopts production of helium-3 as a strawman industrial objective and places enough facilities and infrastructure on the Moon by 2025 to return 1 GWe helium-3 fusion fuel to Earth.

(6) The Mars settlement program moves towards Mars settlement. A robust nuclear electric propulsion system is fielded, with convoy flights by 2015. Mars population reaches 24 by 2025, and the transportation system is capable of increasing Mars population by 24 per opportunity by 2025.
## Three Activity Levels for Architecture Evaluation

<table>
<thead>
<tr>
<th>Minimum</th>
<th>Median (full science)</th>
<th>Industrialization (settlement)</th>
</tr>
</thead>
<tbody>
<tr>
<td><em>Just enough to meet President's objectives</em></td>
<td><em>Meet science objectives of lunar/Mars exploration</em></td>
<td><em>Return of practical benefits to Earth</em></td>
</tr>
<tr>
<td>- Permanent lunar facilities, not permanent human presence</td>
<td>- Human permanence</td>
<td>- Extensive facilities and infrastructure on the Moon by 2025</td>
</tr>
<tr>
<td>- Astrophysics observatories</td>
<td>- Opportunity for lunar geoscience</td>
<td>- Lunar population 30 by 2025</td>
</tr>
<tr>
<td>- Man-tending capability</td>
<td>- In-situ resource technology</td>
<td></td>
</tr>
<tr>
<td>- Explore interesting sites</td>
<td></td>
<td>- Mars population 24 by 2025</td>
</tr>
<tr>
<td>- Three missions to Mars</td>
<td>- Order of magnitude more crew time on Mars</td>
<td>- Capable of increasing Mars population by 24 per opportunity by 2025.</td>
</tr>
<tr>
<td>- Similar to Apollo</td>
<td>- Approaches permanent base (stay time &gt;1 year)</td>
<td></td>
</tr>
</tbody>
</table>
Minimum Program

The minimum program reference averages about 1/2 lunar trip per year and has only three Mars missions. Lunar science facilities are man-tended. Each Mars mission carries two landers (MEVs) for added exploration capability and a measure of rescue capability. Surface stays are about 30 days. Lunar and Mars in-space transportation systems are expendable.
Full Science Program

The full science program reference has about 2 lunar missions per year, to establish permanent human presence on the Moon with adequate supplies and equipment for extensive science and exploration. Lunar oxygen for lunar transportation is introduced about mid-way through the lunar program. Six Mars missions are accomplished, with later missions staying on Mars for more than a year. The Mars missions use multiple landers, as many as four late in the program.
**Full Science Program**

- **First lunar landing**
- **Campsites**
- **First LOR mission**
  - Lunar missions continue at 1 crew & 1 cargo per year
- **First Mars landing**
- **Subsequent Mars landings**

**Technology Advancement**
- NTR engine
- C/D Activities
  - NTR Stage
  - MTV/Hub
  - MEV
- Node ass'y
- Space Operations at node
- Transfer to Mars
Industrialization and Settlement Program

The industrialization and settlement program is very aggressive for both the Moon and Mars. Thousands of tons of industrial equipment are delivered to the Moon, driving lunar cargo trips up to five per year. Lunar oxygen is placed in production as early as possible. One crew trip per year leads to a population of 30 because crew stay times on the Moon increase to several years.

Initial Mars missions use a cryogenic/all-propulsive system because the aggressive nature of the scenario merited an initial Mars mission as early as possible, and the reference nuclear electric propulsion system cannot be ready in time. The NEP missions are operated in a crew rotation/resupply mode, opposition profile, with each crew staying one synodic period (about 2.2 years). The reference scenario evolves to reusable MEVs based on Mars, fueled from Mars resources. Heavy cargo capability is provided, up to 250 t. per opportunity by 2020. The Mars population grows to 24, and by the end of the scenario can continue to grow by 24 or more per opportunity.
Lunar/Mars Program Comparisons

The next two charts compare the lunar and Mars program scenarios in terms of population, cumulative cargo delivered, and flight rate. The lunar population for the minimum scenario is four people for 30 to 40 days about every other year. The Mars population for the minimum scenario is 6 people on each of 3 conjunction missions, with 30 to 40 day surface stays. The full science menu scenario grows to year-long surface stays on conjunction missions. The lunar industrialization program goes to long stay times with indigenous food growth to build population. The Mars proto-settlement program obtains continuous presence by operating the NEP on an opposition-like profile in crew rotation/resupply mode. Later in this scenario, a second NEP is operated to provide two trips to Mars each opportunity.

- These scenarios were the “input” to the manifesting and life cycle cost analyses.
Lunar Program Comparison

Lunar Population

- Industrialization
- Full Science
- Minimum

Program Year

Cumulative Delivered Payload, t

2000 2010 2020

Cargo

- Industrialization
- Full Science
- Minimum

Program Year

Flight Rate

Industrialization
- Full Science
- Minimum

Program Year

Annual Lunar Trips (crew & cargo)

2000 2010 2020
Mars Program Comparisons

Population

Cumulative Delivered Payload, t.

Mars Population

Program Year

2000 2010 2020

Cargo

Cumulative Mars Trips

Settlement

Full Science

Minimum

2000 2010 2020

Flights

Settlement

Full Science

Minimum

2000 2010 2020

Program Year

/STCAEM/grw/4Jan91
Architecture/Launch Vehicle/Node Trends

ISSUES

- Launch vehicle size, shroud size, and lift capacity.
- Node complexity and cost.
- On-orbit assembly complexity
- Number of launches per year
- Development cost
- Per-mission cost

TRENDS FROM ARCHITECTURE ANALYSES

- Large launch vehicle (up to 300 t. lift) does not eliminate on-orbit assembly.
- Keys to on-orbit assembly are (1) design the vehicle to keep it simple; (2) design for automation and robotics; (3) reusable space vehicles to reduce the frequency of assembly operations.
- Advanced in-space transportation technology reduces launch requirements enough that a 100-t., 10-meter shroud launch vehicle is adequate.
- Ultra-large launch vehicle results in high early program costs and is much more costly than advanced in-space transportation technology.
- Evolution and design for evolutionary transitions are the keys to affordable, efficient programs with long-term growth.
Available Options

The facing page is a typical listing of the element options making up a total transportation architecture for SEI missions. The options listed are all candidates for incorporation into architectures. Trade studies have not eliminated any of these options. (The list is representative and not necessarily complete.) The number of options on this chart for each row of options is indicated on the far right. In most cases, any option can be combined with any other set of options. Thus, the total possible combinations number in the millions. It is clear that available future effort can not hope to examine all combinations. This drives us to a strategy for architecture sensitivities analysis, to develop key trends and conclusions from relatively few architecture combinations.
## Available Options

### Advanced Civil Space Systems

<table>
<thead>
<tr>
<th>ETO</th>
<th>100 t.</th>
<th>140 t.</th>
<th>200+ t.</th>
<th>Add prop tanker</th>
<th>No. of options</th>
</tr>
</thead>
<tbody>
<tr>
<td>Node</td>
<td>SSF</td>
<td>Separate</td>
<td>SSF + separate</td>
<td>Self-assy.</td>
<td>Wet tanks</td>
</tr>
<tr>
<td>Lunar mode</td>
<td>Direct</td>
<td>Direct/ lunar ox.</td>
<td>LOR</td>
<td>LOR/ lunar ox.</td>
<td>L2/lunar oxygen</td>
</tr>
<tr>
<td>LTV</td>
<td>Cryo all-prop</td>
<td>Cryo aerobrake</td>
<td>NTR</td>
<td>NEP/SEP cargo</td>
<td>Fully reusable</td>
</tr>
<tr>
<td>LEV</td>
<td>Cryo</td>
<td>Storable</td>
<td>Combined with LTV</td>
<td>Fully reusable</td>
<td>Partially reusable</td>
</tr>
<tr>
<td>Mars mode</td>
<td>2.7 year</td>
<td>1.5 year</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>MTV</td>
<td>Cryo all-prop</td>
<td>Cryo aerobrake</td>
<td>NTR</td>
<td>NEP</td>
<td>SEP</td>
</tr>
<tr>
<td>Mars node</td>
<td>LEO</td>
<td>L2</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>MEV</td>
<td>Cryo</td>
<td>Storable</td>
<td>Combined with LEV</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Total possible combinations 2,799,360
Top-Level Trade Table

The facing page considers mission profile, basing at Mars, and propulsion. Four important issues are central to mission profile selection: crew radiation exposure, crew time spent in zero g, the component of mission risk that increases with mission duration, and the added cost of shortening trip time. At one extreme is the notion, frequently expressed, that a Mars round-trip mission should be completed in a year or less. This is possible with certain advanced propulsion technologies, but at considerably higher cost than for longer trips, as described later in this section of the briefing. At the other extreme, trip time is seen as much less important than minimum mass and cost; conjunction profiles should be used. Crew time in zero g can be minimized by artificial-g spacecraft design. Increase in risk with duration is difficult to quantify. The mission duration issue presently is concerned mainly with cosmic ray exposure.

Crew radiation exposure comes from solar proton events (flares) and galactic cosmic rays, and from mammade sources if nuclear propulsion or power are used. Unshielded energy deposition from GCRs varies from 50 to 100 milligray (5 to 10 rad) per year. The low end of the unshielded range does not constrain Mars mission architectures, but the high end exceeds the present NCRP astronaut radiation guideline of 500 millisieverts/yr (this guideline is for space shuttle and space station missions; no guidelines have been given for Mars missions). It is possible that guidelines will be reduced in the future.

Five profile options are presented. Conjunction fast transfer implies transfers much less than one year. Opposition/swingby trajectories vary from about 440 to about 550 days. Opposition/fast profiles imply 450 days or less, without swingby. The split sprint is a variation on the fast opposition profile in which the MEV and propellant for the return from Mars are sent in advance on a low-energy profile.

If galactic cosmic ray exposure must be controlled, we must either provide shielding on the transfer vehicle or reduce exposure times. Shielding the transfer vehicle habitat dramatically increases its mass, requiring high performance propulsion such as nuclear, or favoring a cycler concept where massive habitats are emplaced on a suitable repeating trajectory and left there. To reduce exposure time, the applicable profiles are: (a) conjunction missions with fast transfers, i.e. less than 180 days, (b) fast opposition profiles, e.g. less than 1-year round trip, and (c) Mars surface rendezvous (Mars direct). The cycler/semi-cycler architectures offer shielding on the Earth-Mars leg, typically 5 months, and provides a 5-6 month conjunction transfer on the return trip. During the long stay at Mars, the crew must be on the surface most of the time unless a shielded Mars orbit habitat is also provided.

Fast-transfer conjunction missions may require orbit basing. A surface rendezvous mission may not be able to achieve the fast return transfer directly from Mars' surface with reasonable vehicle mass, because of the higher delta V required and because the payload launched from Mars' surface is the entire Earth return habitat rather than a lightweight, short-duration crew cab. Available propulsion options become very limited for fast missions. At one year, the only sensible options are NTR splits, where return propellant is prepositioned at Mars on a low-energy profile, or the use of a nuclear gas-core rocket. Below one year, the gas-core rocket quickly becomes the only option.
## Top-Level Trade Table

<table>
<thead>
<tr>
<th>Mission Profile</th>
<th>Propulsion</th>
<th>Basing</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Cryo/All-Prop</td>
<td>Cryo/Aerobrake</td>
</tr>
<tr>
<td>Conjunction Minimum Energy</td>
<td>√</td>
<td>No advantage over propulsive capture</td>
</tr>
<tr>
<td>Conjunction Fast Transfer</td>
<td>Excessive IMLEO</td>
<td>√</td>
</tr>
<tr>
<td>Opposition/Swingby</td>
<td>Same</td>
<td>√</td>
</tr>
<tr>
<td>Opposition/Fast</td>
<td>Same</td>
<td>Excessive IMLEO</td>
</tr>
<tr>
<td>Opposition/Split Sprint</td>
<td>Same</td>
<td>Same</td>
</tr>
</tbody>
</table>

Note 1: NEP flies an opposition/swingby-like-profile but does not benefit from Venus swingby.
Architecture Results for Three Activity Levels

The top-level architecture selection results for the three activity levels are shown on the facing page. For the minimum program, a cryogenic expendable tandem-staged direct mode is the clear economic winner. Its lower development expense causes the operational cost savings for a reusable LOR system to have little payoff. At the median activity level, the reusable system gives about a 5% return on investment (ROI). Our baseline program included lunar oxygen at the median level, but the ROI is estimated only about 3%. At the high lunar activity level, reusable systems and lunar oxygen both have strong payoff, e.g. the lunar oxygen ROI is about 10%.

The minimum Mars program is most economic with cryogenic all-propulsive expendable vehicles on conjunction profiles. The NTR has an ROI less than 2% at this level. If natural environment radiation concerns lead to a conjunction fast transfer or opposition profile, the NTR is the preferred solution with cryogenic/aerobraking as a backup. At the median level, the NTR has a 16% ROI versus cryo all-propulsive. Here also, aerobraking is a backup and SEP comes into the picture as a "dark horse", with about 10% ROI if array costs can be reduced to $100/watt, a tenfold reduction from present costs. At $500/watt, the SEP has a negative 10% ROI, showing the great leverage of array cost. At the high level, electric propulsion is indicated as important, but development costs are a problem unless low-cost SEP arrays can be produced. If electric propulsion costs are too high for a settlement-scale Mars program, the NTR/dash and Mars direct modes are viable options.
## Architecture Results for Three Activity Levels

**Minimum**

- **Lunar:** Expendable

**Mars:**
- Cryogenic all-propulsive
- Unless radiation environment requires reduced trip times; then nuclear rocket or cryo aerobrake conjunction fast transfer

---

**Median (full science)**

- **Lunar:** Start expendable, possible growth to LOR reusable, aerobraking

- **Mars:**
  - Nuclear rocket, conjunction, multiple landers
  - Opposition or conjunction fast transfer options
  - Cryo/aerobraking backup
  - SEP "dark horse"

---

**Industrialization**

### /settlement

- **Lunar:** LOR crew and tandem direct cargo, reusable, with lunar oxygen

- **Mars:**
  - Early cryo/all-propulsive option
  - Electric propulsion for sustained growth (probably SEP)
  - Nuclear rocket/dash or Mars direct/Mars propellant, options for crew rotation and resupply.
Seven Architecture Recommendations

The next seven pages contain our main architecture recommendations with data illustrating key points.
Lunar Architecture

- Begin the lunar program with a tandem-direct expendable system.
  - System can be designed to eliminate on-orbit assembly; one docking or berthing required.
  - The number of development projects is minimized. Offers reasonable expectation of return to the Moon by 2004 under likely funding constraints.
  - Flight mechanics constraints for LOR operations are avoided.
  - Tandem-direct LTV is a starting point for evolution to all other identified lunar architectures.
  - Lunar aerobrake can be tested on the unmanned booster stage without risk to the crew. Stage is otherwise expended.
Lunar Transportation Family
Evolution
Lunar Cryogenic Propulsion

- **Invest in cryogenic storage and management technology.**

  - Without advanced development of a low-boiloff flight-weight cryogenic insulation system, the lunar program may be forced to a storable propulsion system for lunar vicinity operations. Cost impact is billions of dollars.

- **Invest in a 30K-class advanced expander cryogenic engine with 10:1 or better throttling capability.**

  - An advanced expander engine offers about 20 seconds' Isp gain over a modified RL-10; can demonstrate advanced health monitoring and maintainability features essential for Mars missions.
Early Lunar Mode Sensitivity to Isp

![Graph showing sensitivity to Isp for 10 t. and 5 t. payload delivered with crew](image)

- IMLEO, t. for Tandem Direct
- Expendable Mode

Engine Isp vs. IMLEO, t. for Tandem Direct Expendable Mode

- 10 t. payload delivered with crew
- 5 t. payload delivered with crew
Mars Baseline Architecture

- **Baseline nuclear thermal rocket propulsion for Mars.**

  - Nuclear thermal rocket indicated as very economic and flexible over wide range of program activity levels.

  - Nuclear rocket vehicle mass is sensitive to specific impulse. Isp gain for carbide fuels is well worth the technology investment.

  - Development and qualification testing requires proven test facility technology that contains hydrogen effluent and scrubs radioactivity.

  - Nuclear rocket performance permits modest lunar program and significant Mars exploration with about six launches per year of 100-tonne class HLLV.

  - Nuclear rocket baseline offers reasonable expectation of initial Mars mission by 2010 under likely funding constraints.

- **Recommended technology advancement program:**
  - High-performance fuels
  - Full-containment ground test facilities.
Nuclear Rocket ROI Trades

- Nuclear rocket vs. cryo all-propulsive minimum program
- Nuclear rocket vs. cryo all-propulsive minimum program, conjunction profile
- Nuclear rocket vs cryo aerobraking full science program opposition profile
- Nuclear rocket dash profile vs NEP opposition profile, settlement program

Return on Investment, %

50
40
30
20
10
0
Aerobraking Technology

- Accelerate aerobraking technology for Mars aerocapture as backup to nuclear rocket.

- Target decision between the two in the 1996-2000 time frame.

- NTR performance and cost uncertainties, especially test facilities and testing, merit backup.

- Aerobraking needed for Mars landing. Technology challenges less daunting than aerocapture, but merit technology program.

- Aerobraking technology keeps other options open.
  - Conjunction fast transfer
  - Mars direct
  - Cycler orbits
  - NTR-dash profile

- Aerobraking is economic for lunar transportation at >= two flights/year.
Mars Aerocapture Benefit for Conjunction Fast Transfer

**Graph:**
- **Y-axis:** Total Velocity Change (km/sec)
- **X-axis:** Transfer Time (days)

**Legend:**
- **2018**
- **2018**
- **2025**
- **2025**

**Bars:**
- **Cryogenic All-propulsive**
- **Cryogenic Aerobraked**
- **Nuclear Thermal Rocket**

**Bar Chart:** IMLEO, 1000 t

**Notes:**
- D65.15-0262-2
- STCAEM/grw/4Jan91
Program Implementation Architectures Relation to Aerobraking

The facing page indicates uses of aerobraking for the various architectures. As noted, some form of aerobraking occurs in all of the architectures, in particular for Mars landing and for Earth capture on return from lunar missions. In addition, some of the architectures include an Earth crew capture vehicle (ECCV) for direct return of the crew to Earth in cases where, for example, an NEP or SEP vehicle must spiral back down to LEO or in the case of an NTR where the vehicle captures into a highly elliptic orbit.
# Program Implementation Architectures

<table>
<thead>
<tr>
<th>Architecture</th>
<th>Features</th>
<th>Aerobraking Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cryogenic/aerobraking</td>
<td>Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.</td>
<td>x</td>
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<td>Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.</td>
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</tr>
<tr>
<td>L2 Based cryogenic/ aerobraking</td>
<td>L2-based operations; optional use of lunar oxygen.</td>
<td>**</td>
</tr>
<tr>
<td>Direct cryogenic/ aerobraking</td>
<td>Combined MTV/MEV refuels at Mars and LEO. &quot;Fast&quot; conjunction profiles.</td>
<td>x</td>
</tr>
<tr>
<td>Cycler orbits</td>
<td>Cycler orbit stations a la 1986 Space Commission report</td>
<td>***</td>
</tr>
</tbody>
</table>

**Notes:**  
* optional/emergency mode  
** opposition class only  
*** MEV-class crew taxi (not a large MTV)
Aerobraking Flight Test Bed

- Perform aerobrake tests on the LTV booster, to put the technology on the shelf for Mars.

- If the lunar program grows to high activity levels, lunar aerobrake is economically justified.

- A space-assembled aerobrake is needed for Mars landing.

- Aerocapture technology is needed as backup to Mars NTR.
Aerobrake Assembly Test in LEO

Aerobrake Plan

Cross Section

Assembly arm rotates brake as outer panels are installed for easy RMS reach and crew visual contact during operations
SEP as "Dark Horse"

- Designate solar-electric propulsion (SEP) as a "dark horse" for Mars transportation.

- Technology advancement issues:
  - Light weight, high performance, radiation resistant arrays.
  - Automated production technology, $100/watt
  - Robotics technology for constructing SEP and deploying arrays
  - Long-life, high power density, efficient electric thrusters

- If safety precludes operation of nuclear propulsion in low Earth orbit, SEP is the only option more economic than cryo-genic/aerobraking.

- If low-cost array target achieved, SEP is more economic than NEP.

- SEP is the most likely architecture for eventual private sector use for Mars settlement.

- SEP technology has derivative benefits, e.g. power beaming to planet surfaces.
Return on Investment Comparison
SEP versus NTR; Full Science Program

BOEING

Return on Investment, %

SEP at $100/w

SEP at $500/w
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• **Continue the nuclear space power program towards near-term systems applicable to planet surface power.**

• DDT&E and production cost estimates from this study eliminate nuclear electric propulsion (NEP) as a top contender, but are very preliminary.

• As NEP systems are better understood, estimates may come down.

• To keep NEP option open:
  
  • Further studies to better understand the cost of nuclear power systems suitable for electric propulsion.
  
  • Modest funding of high-leverage high-performance power conversion technology.
Mission Risk Comparison

Mission risks were compared in a semi-quantitative way. The methodology is rigorous and quantitative, but reliability and safety estimates for SEI hardware and maneuvers are no more than ballpark guesses today. We made representative estimates with an attempt to be consistent, i.e. the same type of maneuver was given the same number for all cases. Plausible differences were used, e.g. aerocapture was judged higher risk than propulsive capture. Abort modes were included where available.

The facing page shows comparative risks for crew loss and mission loss for several architectures and modes.

NTR shows the least risk because of the propulsive capture advantage, and because a free return abort was assumed, as it was for the cryo/aerobrake. The NTR/dash mode does not permit free return abort or descent abort at Mars, so some mission loss risk turns into crew loss risk. As Mars transportation matures and a safe refuge on the surface of Mars is available, the NTR/dash mode is deemed acceptable. The NTR split sprint mode also exhibits higher risk because of lack of abort modes, e.g. no free return. NEP is shown comparable to, but slightly riskier than NTR. The NEP case is sensitive to the lifetime dependability of the propulsion system; this figure is much more uncertain than NTR reliability. Mars direct has a higher mission loss risk because of its complex automated operations, but the crew loss risk is comparable to the others. The perception of crew loss risk for Mars direct is probably higher than the real risk.
Mission Risk Comparison

- Cryo Aerobrake
- NTR
- NTR Dash
- NTR Split Sprint
- NEP
- Mars Direct

Architecture/Mode

Probability of Loss (no scale)

Mission

Crew
Man Rating Requirements

The facing page describes our recommended approach to man-rating and lists the systems/subsystems for which we believe man-rating is required.
Man-Rating Requirements

Approach

- Ground-based testing wherever possible.
- Use flight program activities to bootstrap, e.g. lunar aerobrake program builds confidence in Mars aerobrakes.
- Flight demonstration of critical functions, e.g. Mars cargo landing, before critical manned use.
- Life demo for long-duration systems before critical manned use, e.g. ECLSS on SSF or lunar surface before manned Mars mission.

Subjects

- Aerobrakes
- Cryogenic rocket engines
- Nuclear rocket engines
- Cryogenic propellant systems
- Attitude control propulsion systems
- Nuclear & solar electric propulsion systems
- ECLSS/TCS
- Crew modules/hab systems
- Vehicle power
- Avionics & Communications systems
- Surface transportation systems

/STCAEM/grw/1Jan91
Nuclear Rocket Man-Rating Approach

A sequence of major tests and demonstrations to achieve nuclear rocket man-rating is shown. Note that two flight demonstration options exist. A decision of which to use depends on whether cargo delivery to Mars is needed before the first manned mission, as would be the case if a conjunction fast transfer and long surface stay is required on the first mission to reduce galactic cosmic ray exposure to the crew.
## Nuclear Rocket Man-Kaung Approach

<table>
<thead>
<tr>
<th>Year</th>
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<tr>
<td>91</td>
<td>Begin fuel form tests</td>
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<tr>
<td>92</td>
<td>Test facility requirements and design approach</td>
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<tr>
<td>93</td>
<td>Reactor design &amp; technology level selected</td>
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<td>Electric furnace fuel tests complete</td>
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<tr>
<td>95</td>
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<td>96</td>
<td>Reactor tests complete; fuel &amp; core design qualified</td>
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<tr>
<td>97</td>
<td>Engine development tests complete</td>
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<tr>
<td>98</td>
<td>Engine qualification test program complete</td>
</tr>
<tr>
<td>99</td>
<td>Mars cargo mission or lunar mission using nuclear rocket</td>
</tr>
<tr>
<td>00</td>
<td>Manned Mars mission using nuclear rocket</td>
</tr>
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</table>
The next three charts present our current recommendations for technology advancement and advanced development, with schedules and funding estimates. The funding level averages about $300 million per year. If we consider the median (full science) program as representative, the technology/advanced development program is about 0.2% of the life cycle cost of the program to 2025, a very modest investment.
# Technology Development Schedules
- Overview -

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<tr>
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<td>▼ AFE flight</td>
<td>△ Lunar engine AD complete</td>
<td>△ LTV design complete</td>
<td>△ Flt. tests comp</td>
<td>△ Final AR&amp;D flt. test</td>
<td>△ Struct. concepts validated</td>
<td>△ LTV Adv. dev./integ. complete</td>
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(~ 2010)
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/STCAEM/jrm/16Jan91
Life Cycle Cost Model Approach

Our basic cost model kernels are parametric cost models. We use the Boeing Parametric Cost Model and the RCA Price models to estimate development and unit cost. The determination of hardware to be costed comes from what architectural elements are needed and from element commonality of the architecture. Program schedules determine requirements and timing for major facilities and for the element development and buy schedules. All of these inputs are used to estimate annual funding for each component of the program, using cost spread functions. The costs are integrated into a spread sheet life cycle cost model to obtain annual funding for complete programs.

The ground rules used in this analysis are indicated on the chart.

The ground rule for use of closed ecological life support (CELSS) and lunar oxygen comes from economics trade studies conducted several years ago through last year.
Life Cycle Cost Model Approach

Ground Rules

- No precursor missions costed.
- NASA contingency not added
- Common element in new application gets 25% delta DDT&E cost.
- No production learning unless production rate > 1 per year.
- Production rates maintained minimum of 1 per 5 years to keep lines open.
- Mission definitions flexible to enable transportation systems to operate at high efficiency.
- All scenarios include closed ecological life support and ISRU for efficiency.
Architectural Cost Drivers

Our investigations of architectures, while preliminary, indicate the importance of cost drivers, in the order listed on the chart. The number of development projects should be minimized through commonality and phased by evolution so that development costs are reduced and are spread over the life cycle of the program, rather than lumped early in the program.

Space hardware for SEI missions is expensive and should be reused if possible. As an example, our unit cost estimate for the Mars transfer crew module is more than a billion dollars. Reuse of this equipment motivates investment in the advanced transportation technology needed to make it reusable.

The third point is that Earth launch mass drives Earth launch cost. Even if Earth launch cost is reduced by ALS-class vehicles, the Earth launch cost is the largest single part of program cost.

The final point is that design and development of systems with mission and operation flexibility enhances commonality and minimizes the risk that changes in mission requirements force new developments or major changes.
Architecture Cost Drivers

- Number of development projects (minimize through commonality)
- System reuse (maximize)
- Earth launch mass (minimize)
- Mission and operational flexibility (maximize)
In-Space Transportation DDT&E Comparison

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Hardware DDT&E in Millions

<table>
<thead>
<tr>
<th>Lunar exp.</th>
<th>Lunar LOR reuse</th>
<th>Cryo all-prop</th>
<th>Cryo aero-brake</th>
<th>NTR 40 MW</th>
<th>NEP $1000/w</th>
<th>SEP 10 MW</th>
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<td>40000</td>
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//STCAEM/grw/4Jan91
Minimum Program Life Cycle Cost Spread

The minimum program life cycle cost spread peaks between five and six billions per year. The deep valley between lunar and Mars peaks indicates that the Mars program should occur earlier in this program. The minimum program involves relatively modest investments in surface systems and falls well below the SEI funding wedge implied by the Augustine Committee recommendations.
Median (Full Science) Program Life Cycle Cost Spread

The median life cycle cost spread peaks at about eight billions per year. With addition of likely surface systems costs, this program probably exceeds the Augustine guidelines during the peak years.

The median program exceeds by a factor of several the science and exploration potential of the minimum program. Lunar human presence grows from an occasional 45 days to permanent presence of six people, and Mars surface time grows from about four man-years to about 30. In other words, a roughly 50% increase in cost leads to about an order of magnitude increase in exploration and science potential.
Median (Full Science) Program Life Cycle Cost Spread
Reduced Early Lunar Program

By deferring major lunar activities, the median program can be brought within the Augustine guidelines. Permanent human lunar presence is delayed until after the Mars DDT&E peak. The early lunar program is like the minimum scenario, i.e. man-tended astrophysics observatories.

Another way to level the funding profile for the median program is to defer Mars by a few years. The reference median program achieves a Mars landing in 2010 (2009 departure). Deferral to about 2016 would probably smooth out the funding profile much as did the reduction of the early lunar program.

Our view was that getting to Mars early was more important than an early buildup to permanent lunar presence. The partially deferred lunar program represented here still achieves astrophysical observatories early, but defers permanent human presence until after the major Mars mission DDT&E is complete.
Industrialization and Settlement Cost Spreads

Our maximum scenario involved simultaneous industrialization of the Moon and progress towards settlement of Mars. As the cost spread shows, this is clearly beyond the funding levels recommended by the Augustine Commission. Both of the premises of this scenario, however, suggest significant private sector involvement.

What is significant in the result presented here is that investment on the order of $100 billions over about 20 years stretches from a plausible public-sector program of science and exploration to a program also involving the private sector for industrialization and settlement. This amount of funding is more than the private sector investment in the Alaska oil pipeline by a factor of a few, and probably less than the private investment in oil supertankers since the closure of the Suez Canal.

The economic potentials of lunar and/or Mars industrialization and settlement are presently not at all understood. We have made some stabs at estimating the costs. We have little or no idea as to the eventual payoffs.
Results of Return on Investment Analyses

The facing page summarizes results of return on investment analyses. (The ROI methodology is explained in the technology and programmatic section of this briefing book.) Results designated "no ROI" had one case always more expensive than the other. An ROI can be calculated only when funding streams cross.

The storable case has very negative ROI because while less (i.e. no) technology money is spent, more vehicle stages must be developed so that the negative cost impact of not doing the essential cryo management and engine technology is large and early. The case for reusable lunar transportation is negative for a minimum lunar program and weak for a median program; it is strong for an industrialization-class program.

The other results were discussed earlier and are included here for completeness.
## Return on Investment Analysis Summary

<table>
<thead>
<tr>
<th>Case</th>
<th>Stor LOR vs cryo direct exp</th>
<th>Reus. LOR vs cryo direct exp</th>
<th>Reus. MEV vs exp MEV</th>
<th>SEP vs NTR $100/w</th>
<th>NTR vs NTR cryo all-prop</th>
<th>NTR dash vs NTR cryo aero brake</th>
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<td>Reus MEV higher LCC</td>
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Strategy for Architecture Synthesis

The strategy we have adopted is illustrated on the facing page. First, we examined propulsion systems options through trade studies to understand how they work and to define preferred configuration operating modes. Secondly, based on the knowledge gained through these trade studies we chose a set of architectures using combinations of systems and modes, paying attention to integration compatibility, evolutions and commonality. Third, we will compare and trade architectures over a range of scopes and obtain important sensitivities and understand how architectures respond to program scope. We expect this analysis to lead to preferred architectures for various scopes. The final step is to conduct trades within the winning architectures to make further improvements.

All of this is guided by knowledge of the architecture cost drivers described earlier and by the knowledge gained on how systems work together, from the trades conducted within individual propulsion systems.
Strategy for Architecture Synthesis

Advanced Civil Space Systems

1. Conduct trades within winning architectures to improve them

Criteria:
- Risk
- Performance
- Cost
- Flexibility

2. Compare and trade architectures over a range of program scopes to obtain important trends and sensitivities

Criteria:
- Ability to satisfy program goals & schedules
- Cost

3. Synthesize candidate architectures using effective combinations of systems and modes

Criteria:
- Integration
- Compatibility
- Evolution
- Commonality of hardware & technology

4. Conduct trades on systems to find preferred configurations and operating modes

Criteria:
- Risk
- Performance
- Cost
- Flexibility
Architectures Synthesis vs Mission/System Analysis

The facing page compares this approach to the traditional top-down systems engineering approach. The traditional approach shown on the right, starts with program goals, establishes mission requirements through trades, and continues to lower levels. As usually conducted, the traditional approach is faced with the great number of possible combinations noted earlier. The usual outcome is that requirement decisions are made and systems selected without trade studies.

The synthesis technique, on the left, attempts to avoid this problem by a combined top down/bottom up approach. It is similar to a classical optimization problem.

Optimization deals with infinite numbers of paths that satisfy boundary conditions. Optimization is a technique for generating only optimal paths. Any path that satisfies the boundary conditions is the sought optimal path.

Nothing quite as rigorous can be done in architecture synthesis. However, by bottom up trades, assembling systems into "good" candidate architectures, and matching with ranges of program scope, we may come close. The key is knowledge we obtain on what works well what things are compatible and combine well to satisfy mission requirements.

The last step is to conduct trades and analyses such as life cycle cost to identify preferred architectures, apply criteria derived from national goals program goals, to select among preferred architectures.

The dotted line indicates that one could then enter the traditional analysis flow with preferred architectures and their associated requirements and mission profiles, to further refine systems through systems engineering.
Architecture Trade Flow

The facing page shows the low level system mission and operations trades that have been conducted or are being conducted for our seven architectures to represent the range of possible architectures for the SEI mission. Most of the trade areas have been presented in this briefing or have been presented in earlier briefings. The knowledge base in this area is fairly complete except that only very preliminary analyses have been done for the cryogenic direct mode and for cycler orbits. When these two options are completed we will be ready to finish up the architecture analysis.
## Architecture Trade Flow

### ADVANCED CIVIL SPACE SYSTEMS

<table>
<thead>
<tr>
<th>Cryo/Aero-braking</th>
<th>Nuclear Electric (NEP)</th>
<th>Solar Electric (SEP)</th>
<th>Nuclear Thermal Rocket (NTR)</th>
<th>L2/Lunar Oxygen</th>
<th>Cryo/Aero-braking Direct</th>
<th>Cyclers Orbits</th>
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<tbody>
<tr>
<td>- Mission design</td>
<td>- Mission design</td>
<td>- Mission design</td>
<td>- Mission design</td>
<td>- All-propulsive conj. option</td>
<td>- Performance vs. separate MTV/MEV</td>
<td>- Mission design</td>
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<td>- Reuse</td>
<td>- trip time</td>
<td>- trip time</td>
<td>- Isp and T/W sensitivity</td>
<td>- Lunar oxygen benefits</td>
<td>- Sensitivity to propellant choice</td>
<td>- Feasibility of high Mars encounter velocities</td>
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<td>- Aerobrake</td>
<td>- gravity assist</td>
<td>- gravity assist</td>
<td>- Reuse</td>
<td>- Integration of lunar &amp; Mars ops.</td>
<td>- Advanced propulsion for LEO-L2 operations</td>
<td>- Design of &quot;taxis&quot;</td>
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<td>- Redundancy mgmt.</td>
<td>- Assembly/ deployment of large space structure</td>
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**For all:** Overall configuration; key subsystems performance; integration compatibility; operations analyses
Architecture Evaluation Approach

Early Mars

- Cryo/All-Propulsive
- Cryo/Aero-braking
- Nuclear Thermal Rocket (NTR)
- Cryo/Aerobraking Direct

Least early cost
Least early risk
Early Mars readiness

Late Mars & Mars Evolution

- Nuclear Electric (NEP)
- Solar Electric (SEP)
- L2/Lunar Oxygen
- Cyclers Orbits

Solar array cost vs. nuclear cost
Retain all vs. retain most of interplanetary energy

Benefit vs. cost vs. risk vs. high performance, all vs. program scope
Mars Summary

- More than 20 beneficial modes identified.

- Early Mars: Cryo all-propulsive (CAP), ECCV, conjunction;
  NTR all-propulsive, conjunction or opposition;
  Cryo aerobraking opposition, ECCV;
  (possibly) Direct with Mars oxygen.

- High performance, late Mars or evolution:
  SEP or NEP;
  ISRU, moon or Mars or both;
  Combinations.

- Efficiency range 10:1 measured as RMLEO (resupply mass LEO).

- Reusable MEV/Mars propellant has significant leverage for
  high-performance options.

- Earth Crew Capture Vehicle, an Apollo-like capsule used for Earth entry and landing
  or aerocapture to LEO. The rest of the vehicle is expended.
Comparison of Propulsion Options

Resupply Mass to LEO, t.

- Analyzed
- Estimated by scaling eq.
- Estimated by comparative trends

Conjunction  
Opposition
Conjunction vs. Opposition Mars Profiles

Opposition Advantages

- Shorter overall trip time, by at least a year.
- Transfer vehicle usually returns in time to be reused on next opportunity.
- Enables crew rotation/resupply mode with synodic period stay time.

Conjunction Advantages

- Lower energy; significantly less RMLEO unless very high Isp available.
- Venus swingby complexity not necessary.
- Long stay times at Mars.
- Shorter transfer times.
- Elliptic parking orbits can be optimized.
Reusable MEV Sensitivities

ADVANCED CIVIL SPACE SYSTEMS

BOEING

Direct Return To Earth

Direct Payloads
40 t. landed
40 t. return, except as noted

Mars orbit payloads
30 t. landed
5 t. return

Ref. MEV (expendable)
Mini MEV (expendable)
Direct Ref (Mars propellant)
Direct
25 t. Cargo (65 t. landed)
20 t. return payload
10% Brake
Methane 5% tank
Meth 20 t. Ret.
Meth 4% tank
Mars Orbit 1-sol
Mars Orbit 10-hr
Mars Orbit 10-hr 7:1 mixture ratio

15% Aerobrake direct
10% Aerobrake from Mars orbit
8% Tank fraction for O2-112
II. Requirements, Guidelines and Assumptions
Reference and Alternate Missions

Note: Contains material formerly in Mission Analysis.
Mission Analysis

A reference mission profile for Mars transfer was provided by MSFC, called the Level II reference case (year 2015 opposition opportunity). We investigated this profile for other opportunities in other years and did not limit ourselves to opposition mission only. An alternative mission profile is to use a direct transfer to Mars, refueling on the Martian surface, and direct return to Earth orbit (Mars Surface Rendezvous). A third alternative is to use the Earth-Moon Lagrange point two (L2) as a departure and return node.

The reference mission profile for the year 2015 depart on May 22nd of that year and has a 30 day stay time at Mars. The total mission duration is 565 days. A 2016 profile has a shorter overall time, 434 days, but adds about one kilometer/sec to the departure velocity change (ΔV) relative to the 2015 mission. The stay time at Mars is held to 30 days.

Mission opportunities from 2010 to 2024 are tabulated. A plot of departure date versus outbound trip time for the 2013 opportunity with a 400 day stay is typical, showing a single optimum combination that minimizes mission ΔV. The other orbital characteristics than epoch of departure have some effect on mission velocity requirements such as capture and departure S-vector positions, GN&C maneuvers, etc. They have been investigated in as great a detail as the depth and length of the contract allows.

Optimum departure vectors indicate that the ability of the engines to be capable of multiple burns and therefore do broken plane trajectories to the declination launch asymptote will be a requirement. Arrival conditions at Mars for capture orbit parameters such as periapses location and lighting (capture and land in light), impacts of true anomaly and parking orbit period on the relative position of the S-vector (departure vector) for abort and departure capability and into the characteristics of the aerobrake itself for GN&C, landing and crossrange capability.

In the reference mission, the excursion vehicle and transfer vehicle separately capture into Mars orbit. To allow a one day spacing between the captures, a velocity difference must be generated between them. To keep this ΔV low (under 100 m/s), the separation should occur about 50 days before Mars arrival.

Guidance, navigation, and control analyses can be done at different levels of detail from closed form approximations to full 6-degree of freedom simulations. To date 3 degree of freedom analyses with variable atmosphere has allowed assessment of the errors induced by a variable atmosphere. Guidance laws are being investigated.

Aeroheating analyses were performed on the medium (L/D=0.5) and higher (L/D=1.1) lift brake concepts. With a fixed exit velocity target, flying inverted (negative L/D) extends the time for the maneuver. This is because the maneuver is from a hyperbolic velocity. A zero lift trajectory would rise quickly back out of the atmosphere. Negative lift 'holds down' the vehicle, extending the time for the aeromaneuver, thus lowering the heating rate. The heat rate itself varies widely depending on the analysis method. This is an area that requires more detailed investigation in the future. Either method leads to peak temperatures at the stagnation point in excess of 2000K for hyperbolic excess energies (C3) of over 30 km²/sec². Aerodynamic loads were estimated over the brake surface, and two structural concepts were examined. The first was a spar framework, the second was a truss framework.

A summary of this work is given below:
This summary addresses the aerobrake analyses categorized as geometric configuration for capture and landing, Mars atmosphere knowledge uncertainty impacts on GN&C, design configurations for reducing heating rates and loads, landing flight mechanics for range and crossrange requirements, structural techniques for reducing weight, and integration of technology to meet overall mission goals. The aforementioned categories will be covered in four sections: Aerocapture, Heating, Structure, and Ascent/Descent.

Aerocapture - Critical GN&C related aerocapture issues are line-of-apside control and apoapsis altitude control. Aerocapture analyses results included in this summary show the following:

* Asymmetric roll with a finite rate provides improved line of apsides control.
* A guidance system designed for a low density atmosphere needs to be optimized for other atmospheric conditions.
* Using MarsGram, a one sigma density change results in a large difference in density variation between day and night.
* The guidance system (as related to aerocapture exit conditions) is more affected by large (wavelength > 1000 km) horizontal sine wave density variations.
* A larger vertical wavelength (on the order of 20 km) sine wave density induces a lesser error than a smaller vertical wavelength (on the order of 5 km) sine wave density.

Heating - Mars aerocapture heating analyses results are given for stagnation point heating and for some choices of surface heating. Heating analyses results included in this summary indicate the following:

* For the Mars aerocapture MTV, the stagnation point heating rate resulting from averaged lift-down L/D is lower than the heating rate for average lift-up L/D.
* Under similar conditions, the heating loads follow the same trend as the stagnation point heating rate.
* Along the center streamline of the hyperboloid aerobrake the predicted radiative heat transfer rate at Mars using the Park method is approximately two times that using the Tauber-Sutton method.
* The total heating rates at the stagnation point with Park(146 w/sq cm) and Tauber-Sutton (80 w/sq cm) are higher than the near term (1993) radiative material capabilities of approximately 70 w/sq cm.
* For an averaged L/D = 0.5 the stagnation point heating rate for Mars aerocapture is 146 w/sq cm; Earth aerocapture heating rate is 172 w/sq cm.
* The local Reynolds number along the aft streamline of the 30m body does not exceed 10E6.

Structures - Structural analyses results demonstrate weight savings and strength improvements through advanced composites application and through spar design advantages. Included structural analyses results depict the following:
Spar and truss configurations were developed for the 30 meter aerobrake concept.

For the spar configuration and with current technology, the (81 mt payload) weight estimate is 41.5 klb and the MTV (153 mt payload) estimate is 66.3 klb.

Improved material characteristics (200 ksi vs 105 ksi span strength) reduces configuration weight by greater than 15%.

Mass savings of 30% may be achieved by improved spar design and advanced materials characteristics.

The truss configuration provides a 15% weight savings compared to the spar configuration.

Ascent/Descent - No ascent related information is discussed in this version of the IP&ED; a forthcoming update will contain a discussion of ascent related data.

Descent trajectory analyses results point to L/D requirements related to landing site accessibility issues. Included descent trajectory results include the following:

* For MEV with L/D = 1 and descent inclination of 45 degrees, a displacement in latitude of 30 degrees may be achieved.

* An increase in L/D from 0.5 to 1.0+ extends the range by approximately 50%.

* An aeroflare reduces the ideal delta velocity required for landing by 200 to 300 m/sec (L/D = 1).

* Cross range is a function of L/D with atmospheric density and dust concentrations affecting the results

Issues with large aerobrakes such as these center around on-orbit assembly and inspection, functions which consume many man-hours for the Space Shuttle on the ground. The Shuttle has the only reusable aerobrake with repetitive use and accessible data. Another issue is selection of the landing site. If the landing site requires an extensive plane change, the L/D is higher, which ripples through the packaging of the lander and the weight of the lander back to Earth launch requirements. On the other hand, using an arrival orbit tailored to the landing site also has an impact on propulsion requirements, thence to Earth launch mass. Thus selection of a landing site is required early since it affects the whole design in a complex way. Some candidate sites are listed. Either a reference site or a requirement to meet a range of sites up to some level of difficulty (for example any site less than +5km altitude and <70 degrees latitude) needs to be given as an input requirement for further analyses.
Mars Transfer Operations

Mars transfer operations for the reference system are illustrated here. The opposition profiles normally include a Venus swingby either going to Mars or returning to Earth. Occasionally, a Venus swingby may be used each way. (The reference 2015 mission uses an outbound swingby; there is also an alternate inbound swingby profile for this opportunity.) Venus swingbys are normally unpowered and there are no operational events at the swingby. The nominal mission sequence is as follows:

1. Reference Cryo/AB Mars vehicle leaving Earth orbit.
2. MEV/MTV separate 50 days from Mars.
3. Unmanned MEV captures into Mars orbit 1 day prior to MTV.
4. MTV/MEV rendezvous and berth in Mars orbit.
5. Crew transfers from MTV to MEV.
6. MEV descends to the surface of Mars.
7. MAV ascends from surface, leaving descent stage.
8. MAV/MTV berth in Mars orbit.
9. Crew transfers from MAV to MTV.
10. MAV left in Mars orbit.
11. MTV departs from Mars toward Earth.
12. MTV captures in LEO, or crew returns to Earth's surface in ECCV.

Acronyms: AB - aerobraking; MEV - Mars Excursion Vehicle; MTV - Mars transfer vehicle, includes trans-Mars injection stage as well as transfer propulsion and hab; MAV - Mars ascent vehicle; LEO - low Earth orbit; ECCV - Earth crew capture vehicle (like an Apollo command module).
Reference Chemical/Aerobrake Mission Profile Schematic

Aerobraking Earth capture into LEO

Earth

Mars

Trans Earth Injection departure burn

Rendezvous & dock

Separation maneuver 50 days out

Assemble & depart from LEO

Drop TMI stage

2 burn departure

STCAEM/bac/12jun90
Mission Schematic for Mars Direct System
(Conjunction Profiles, Refueled at Mars)

ADVANCED CIVIL SPACE SYSTEMS

BOEING

Delta V Mars to Earth (m/sec):
Ascent to circular 4100
Injection C3 = 10 2373
Earth aerocapture 200
Reserve 120
Total 6793

Transit Earth-to-Mars-landing (can use parking orbit at Mars if needed)

No Mars orbit operations

Delta V LEO to Mars landing (m/sec):
TMI (C3 = 20) 4100
Midcourse & aerocapture 200
Landing 1500
Reserve 115
Total 5915
The use of the L2 libration point as a transportation node was originally suggested by the Farquhar and later by Keaton. The L2 node has three advantages: (1) the delta V from L2 to Mars is much smaller; (2) the transportation node is not in the low Earth orbit debris environment; (3) the L2 node is a suitable location for using lunar oxygen in Mars mission systems.

The L2 node scheme involves a network of transportation from the Earth to L2 and back, from L2 to the lunar surface and back, and L2 to Mars and back. The facing page indicates the delta V's for each leg of the transportation network.

Transportation to the surface of the moon via L2 is efficient when lunar oxygen is used for the LV, as efficient as the low lunar orbital node with lunar oxygen. (The LV becomes like the LTV in size.) The disadvantage of the L2 node is that substantially greater lunar oxygen production is required compared to the low lunar orbital node. The advantage is that any site on the lunar surface is accessible any time and return to Earth is available at any time.

Delivery of lunar oxygen to the L2 node by rocket is surprisingly efficient.

Transfer from L2 to Mars uses dual powered gravity assist, at the Moon and the Earth. This means that launches to Mars are limited to times the Moon is at proper location. The launch window problem is very similar to launching from a Earth orbit. In either case the launch window of conjunction missions, multiple chances are available. The return from Mars uses...
Earth-L2-Mars Mission Profile Schematic

**L2 to lunar surface:**
2950 m/sec

**L2 to L2 (TLI):**
3200 m/sec

**L2 to Earth orbit:**
530 m/sec (aero)
3500 (all prop)

**Earth orbit to L2:**

**L2 to Mars:**
1000 - 1500 m/sec conjunction profile;
2000 - 4000 m/sec opposition profile.

**Mars arrival and departure:**

Earth swingby to Mars:
Depart L2 - 330 m/sec
Earth swingby typ. 1200 m/sec

(sqr(115.9 + C3) - 10.473, km/sec)

(sqr(23.487 + C3) - 4.625, km/sec, plus orbit alignment penalties);
value 4.625 is perigee velocity for 250 km x 1 sol (24.6 hr) orbit
Reference 2015 Mission Profile

Depart Earth
5/22/15
C3 = 16.628
DLA = 58.5°
Δ V = 4281

Venus Encounter
10/19/15

Earth Return
12/7/16
Vhp = 8.78
LVI = -18.1°

Arrive Mars
4/21/16
Vhp = 6.97

Depart Mars
5/21/16
C3 = 30.69
Δ V = 3400

Trip Time = 565 days
Boeing Case #2

Depart Earth
2/25/16
C3 = 10.3
DLA = -35°
Δ V = 4082
including plane change, window, and g loss

Venus encounter
3/10/17

Earth return
5/5/17
Vhp = 7.14
LVI = 17.2°

Arrive Mars 7/31/16
Vhp = 6.775

Depart Mars 8/31/16
C3 = 40
Δ V = 4300
including plane change and apsidal misalignment penalties.

Trip time = 434 days
## Mars Trajectory Data

<table>
<thead>
<tr>
<th>Opportunity (optimized runs)</th>
<th>Earth Dep C3</th>
<th>Mars Arr. Vhp</th>
<th>Mars Departure ΔV</th>
<th>Earth Arrival Vhp</th>
<th>Mars Orbit Inc. (Deg)</th>
<th>Earth launch DLA</th>
<th>Mars Arr. LVI</th>
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Level II Reference: ΔV Earth Departure = 4281 m/sec  ΔV Earth Arrival = 6278 m/sec (at LEO)
Data From MASE  ΔV Mars Arrival = 3949 m/sec  ΔV Mars Departure = 3400 m/sec
## Trajectory Information for Mission Opportunities

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<th>C3 Earth Departure</th>
<th>Vhp Mars Arrival</th>
<th>C3 Mars Departure</th>
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<th>Apoapsis Radius (km)</th>
<th>Periapsis Radius (km)</th>
<th>Eccentricity</th>
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Orbit Insertion Lighting Angle, Latitude, and Approach Turning Angle for 2010 to 2025 Mars Mission Opportunities

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<th>Periapsis Latitude (°)</th>
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Data generated by the PLANET program, property of the Boeing Company.
2013-400 Day Stay -- 900 Days Round Trip

2013 Conjunction minimum round trip $\Delta V$ is approximately 11.5 km/sec with an outbound/inbound trip time of approximately 240 days.
900 Days Round Trip

ADVANCED CIVIL SPACE SYSTEMS

Contour $\Delta V$

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$\Delta V_{min} = 11$

Outbound Trip Time (days)

Julian Date (245XXXX)
Conjunction Class Minimum Energy and Fast Transfer Missions

Shown in the next several charts are the data comparing the results of using a "fast transfer" versus a minimum energy (minimum ΔV) mission for the 2025 time frame. As shown a price in ΔV costs at all stages of the trip must be paid for the reduction of time spent in transit and the reduced risk of crew exposure to Galactic Cosmic Rays. This price will be reflected in the IMLEO of the vehicle.

For a range of trip times and launch dates, contour plots of total ΔV for both cases were generated. It is apparent from the data that the fast transfer missions have narrower launch opportunities at a higher total ΔV price but with a transit time of half or less than a normal mission.
Minimum Energy and Fast Transfer

- Depart Mars 11/16/26 C3=42.6
- Depart Earth 11/15/24 C3=44.3
- Arrive Mars 4/15/25 Vmp=6.45
- Depart Earth 4/15/27 Vmp=7.58
- Return Earth 6/14/27 Vmp=3.51
- Conjunction Minimum Energy 180-day Mission 300-day Transfers

- Depart Mars 9/2/26 C3=11.4
- Arrive Mars 7/29/25 Vmp=2.5

D615-10026-2
Delta-Velocity Contour
Typical Conjunction Mission

Analysis Tool-Boeing's PLANET
2025 Typical Conjunction Mission
1000 Day Total Trip Time
400 Day Stay at Mars

km/s
1  5.3
2  6.0
3  7.0
4  8.0
5  9.0
6 10.0
Conjunction Fast Transfers Preliminary Delta Velocity Trends

This is a comparison of the total required ΔV versus transfer time one way for aerobraked and all-propulsive vehicles. This was done for two years; the fairly "easy year" of 2018 and the "hard year" of 2025. This shows both the effect of the range of years and the effect of using an aerobraked vehicle. The aerobrake shows its advantage in the reduction in ΔV that must be provided. The advantage in going in 2018 is evident as long as the total transfer time is less than 250 days, beyond that the advantage is lost. However this data has not been optimized for the transfer time, and only shows the relative advantage of using an aerobrake.
Preliminary Delta Velocity Trends

Not optimized for trip time or stay time

- 2018
- 2018
- 2025 all-propulsive
- 2025 aerobraked

Total Delta V (km/s)

One Way Transfer Time (days)
Conjunction Fast Transfer Optimized Data

This set of data were optimized for a fixed total transfer times for the 2018 "easy" opportunity and the 2025 "hard" opportunity. This information is applicable to an NTR or Cryogenic All-Propulsive mission. The data shows the same trend as in the data in the preceding chart of all-propulsive curves. If the data is optimized for a fixed transfer time there is still an advantage in going in 2018 for missions with total transfer times less than 550 days. Total transfer times of less than 300 days is possible with an increased price in total ΔV.
Optimized Trip Time and Stay Time for Fixed Transfer Time
All-Propulsive

![Graph showing optimized trip time and stay time for fixed transfer time for all-propulsive systems. The x-axis represents total transfer time in days, and the y-axis represents total delta velocity in km/s. The graph compares DEI V 2018 and DEI V 2025.]
Losses For Two-Burn Trans-Mars Injection

This analysis is a look at the losses in finite burn ΔV for the start altitude required for a second TMI engine burn using the class of engines considered for lunar case main engine use. This shows that it is possible to use these engines to reach the required C3 energy for Mars transit, with more than one TMI burn (broken plane trajectory).
Losses for Two-Burn Trans-Mars Injection
Injection C3 - 20 km2/sec2  T/Wo 0.069  Isp - 481

![Graph showing total finite burn loss against start altitude for 2nd burn, Earth Radii.]

C3s at end of first burn
-25
-35
-30

Start Altitude for 2nd Burn, Earth Radii
0.2  0.3  0.4  0.5  0.6

Total Finite Burn Loss
460  455  450  445  440  435  430  425  420  415  410
Separation $\Delta V$ vs. Days From Mars for MEV One Day Early Entry Arrival

Before the MTV vehicle arrives at Mars, it must separate with the unmanned MEV, which will aerocapture first, before the manned MTV. The difference between the aerocapture of the unmanned MEV and the manned MTV will be one day. To insure this separation, a separation maneuver must be done before arrival. The cost of this separation in $\Delta V$ for is dependent on when it is done. This curve shows that the minimum time from Mars arrival with the minimum required $\Delta V$, (which occurs at the knee of the curve) is 50 days out from Mars. This is for the Level II Reference mission.
Capture Trajectory Paths

These are pictorial descriptions of the aerocapture arrival inclination for the identified missions. This inclination changes, sometimes drastically, from mission to mission. When the object of the mission is to land at an established base site or a site off the track of aerocapture, either a plane change is required or the need for cross-range capability is establish.
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Performance Parametrics

Note: Contains material formerly in Mission Analysis
2010 Conjunction
S Vector and True Anomaly vs Parking Orbit Period

Minimum departure delta V occurs when the S vector is in plane and departure true anomaly is close to periapsis.
2010 Conjunction
S Vector and True Anomaly vs Parking Orbit Period

ADVANCED CIVIL SPACE SYSTEMS

20 Degree Inclination

40 Degree Inclination

60 Degree Inclination

70 Degree Inclination

STCAEM/grw/31May90
Minimum Mars departure AV of approximately 1.2 km/sec occurs for parking orbits with inclinations of 30° and 60° and period of 9 hours each.

Periapsis lighting angle is adequate for the parking orbits with inclinations of 30° and 60° and period of 9 hours each.

Minimum S vector out-of-plane occurs when the true anomaly at departure is closest to periapsis and the orbital period is 9 hours.

Periapsis latitude for a 30°, 9 hour parking orbit provides landing coverage for landing sites between 38° to 50° north latitude; periapsis latitude for 60°, 9 hour parking orbit provides landing site access to landing sites between 5° south to greater than 20° north latitude.
## Analysis Type

- Closed-form zero lift approximation; fixed exponential atmosphere
- Fixed-lift integrated trajectories; 2-DOF; fixed tabulated atmospheres
- Modulated lift integrated trajectories; 3-DOF or 3-1/2 DOF; fixed tabulated atmosphere
- Modulated lift integrated trajectories; 3-DOF or 3-1/2 DOF; variable atmosphere
- 6-DOF integrated trajectories with simulation of vehicle flight control system; variable atmosphere

## Results Obtained

- Depth of penetration versus ballistic coefficient and entry velocity
- Corridor height and g level vs. available L/D and entry velocities; entry conditions
- Trajectory designs for aerocapture, considering vehicle lift modulation capability and rates
- Development of guidance schemes and laws; assessment of errors induced by atmosphere unpredictability
- Accurate assessment of vehicle capabilities for aerocapture; detailed design requirements for aerobrakes and flight control systems
This shows another example of the trajectory design, using the OPTIC code. We are using two different GN&C codes to cross-check results. OPTIC, developed by Boeing-Seattle, optimizes with constraints. The other, AEROPASS, developed by Boeing-Huntsville, optimizes switch points and exerizes guidance laws. Constraints must be represented by penalty functions with this routine.
Guidance Navigation & Control

LIFT MODULATION VIA BANK ANGLE MOD.

< Guidance control through the Mars atmosphere is done by controlling bank angle in a "slalom course" motion

This is the resultant inclination, right ascension angle and argument of periapsis (position with respect to the planet) for the "slalom course" run

< comparison of the atmospheric deviation obtained with the MARSGRAM and Optic codes
Guidance Schemes for Aerocapture

The facing page identifies the requirements for aerocapture GN&C, summarizes two approaches to adaptive guidance, and illustrates a gain-scheduling scheme presently under investigation. We also plan to investigate the real-time re-optimization scheme.
Guidance Schemes for Aerocapture

- Requirements - minimize changes in inclination and line of nodes. Attain desired line of apsides and apoapsis altitude.

- Maximum performance - redesign trajectory optimization and constraints every few seconds, but requires very high computer performance.

- Less performance but may be adequate - use an adaptive gain scheduling scheme that adjusts to experienced atmosphere conditions.
Orbit Correction Analysis

The next three pages illustrate a method of correcting exit conditions with two burns, and show some preliminary results of calculations of the delta V required for each burn as a function of the magnitude of exit errors.
Orbit Correction Analysis

- Burn at V1 will - align the apsides
  - set the apoapsis height to the desired final orbit height

- Burn at V2 will - raise periapsis height
  - correct inclination errors

- State vector errors are:
  - flightpath angle
  - crosspath angle
  - velocity magnitude

- Position Vectors errors are:
  - true anomaly
  - orbit plane
Effect of Burn Location on ΔV Penalties

Velocity Magnitude Error = 0.95
ΔV vs. True Anomaly

Flight Path Angle Error = 1 Deg
ΔV vs. True Anomaly

Cross Path Angle Error = 1 Deg
ΔV vs. True Anomaly

First Delta V + Second Delta V

Delta V = km/sec

True Anomaly (Degrees)
Effect of Burn Location on ΔV Penalties

Page 2

True Anomaly Error = 1 Deg

DELTA V VS. TRUE ANOMALY

ΔV = km/sec

First Delta V + Second Delta V

Position Plane Error = 1 Deg

DELTA V VS. TRUE ANOMALY

ΔV = km/sec

First Delta V + Second Delta V
The delta velocity is independent on the true anomaly at first burn but is dependent on the flight path angle error. The two flight path angle errors examined were one degree and five degrees. Even though the delta V's for the different flight path angles are significantly different, the delta V for the second burn is not dependent on the error in the first burn. The total correction delta V is minimized by performing the first burn close to the perigliss where the true anomaly is zero.
Flight Path Angle Error on Delta V

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Delta V (km/sec)

0.45

0.4

0.35

0.3

0.25

0.2

0.15

0.1

0.05

0

0

20

40

60

80

100

120

True Anomaly at First Burn (Degrees)

1st AV, Error = 1°
2nd AV, Error = 1°
1st AV, Error = 3°
2nd AV, Error = 3°

G GRM/211895/AEROBRAKE STUDY/DISK 3/3165 0/10:00A

203 D615-10026-2
In the early part of the trajectory, the roll angle is set by the gravity
loading which is an indication of the type of atmosphere encountered. In
the early to mid portion of the trajectory, the roll angle is used
to minimize the flight path angle error which is based on the difference
between the actual flight path angle and the predetermined optimum flight
path angle. In the latter part of the trajectory the roll angle is used
to minimize the line of apsides error. Examples of guided trajectory
profiles are provided for low and high density atmospheres.
Guided Trajectory Examples

COSPAR low-density atmosphere

COSPAR high-density atmosphere

Roll Angle

Path

Degrees

0

0

0

0

0

0

0

Time, sec.
For the hyperboloid vehicle, it is illustrated that the constraint can be met with an entry flight path angle of approximately 0.75°. This is illustrated for the COSPAR low atmosphere.
Parking Orbit Constraint Errors

Advanced Civil Space Systems

Cospar - Low, Dia = 0.004

Entry Flight Path Angle (Deg)

Orbital Elements (Deg)

Inclination
RAAN
AROPIN

Linear Range
Mars Aerocapture Trajectory - Finite Roll Rate

This figure shows the effects of finite roll rates on the trajectory design. In going from left to right and back, the lift vector may be rolled over the top, or under. Because this perturbs the vertical path, the rest of the trajectory design must compensate by going a little deeper (roll over) or a little less deep (roll under). These results show that the effect on the vertical path is less with the roll under, and that the maximum deceleration is less, leading to a clear preference for "roll under".
Mars Aerocapture Trajectory - Finite Roll Rate

- COSPAR low-density atmosphere
- Fixed L/D 0.5
- Entry path angle -10°
- Approach C3 50 km/sec
- M/CdA 400 kg/m
Mars Aerocapture Trajectory Design Approach

The trajectory design approach is tailored to a roll-only control scheme. Excess lift is dissipated by veering the trajectory to the left and to the right in a dog-leg or "slalom" maneuver. The illustration on the left shows a nominal symmetric design, with roll angles of 107° (0° is straight up). This applies a net vertical lift coefficient of about -0.15. The asymmetric design on the right enables control of the line of apsides, so that the range of atmospheres represented by the COSPAR low and high density atmospheres can be navigated, from the same entry conditions, to realize the same capture orbit, within reasonable delta V budget for post-exit correction. For the low-density atmosphere, the roll angle is greater during penetration than during exit. For the high-density atmosphere, the reverse is true.
Mars Aerocapture Trajectory Design Approach

- COSPAR low-density atmosphere
- Fixed L/D 0.5
- Entry path angle -10°
- Approach C3 50 km/sec
- M/CdA 400 kg/m

Basic design is symmetric

Asymmetric roll program enables line-of-apsides control
Mars Aerocapture - Guided Trajectory Examples

This shows preliminary results of trying the gain-scheduling scheme. Gains have not been optimized. The same entry conditions were used with the high and low-density atmospheres. The guidance scheme gave good results on all exit conditions except line of apsides, and fair results for that parameter.
Mars Aerocapture - Guided Trajectory Examples

**COSPAR low-density atmosphere**

**Roll Angle**

- Degrees
- 0 to +150
- 0 to -150

- Time
- 0 to 300

**Path**

- Degrees
- 0 to +10
- 0 to -10

---

**COSPAR high-density atmosphere**

**Roll Angle**

- Degrees
- 0 to +150
- 0 to -150

- Time
- 0 to 300

**Path**

- Degrees
- 0 to +8
- 0 to -10
"Slalom Course" Maneuver Profiles

This simulation, with the OPTIC code, examined the effects on a trajectory design of typical atmosphere density variations predicted by MARS-GRAM. The most significant effects were a significant reduction in exit velocity and a large rotation of the line of apsides. No adaptive guidance was simulated. The result shows a clear need for adaptive guidance.
Atmospheric Entry Conditions for "Slalom Course" Maneuver
Atmosphere Entry Conditions for "Slalom Course" Maneuver

An additional display of the trajectory design is shown here. Corridor height parametrics are on the left. A typical trajectory profile for a relatively dense MARS-GRAM atmosphere is on the upper right. Typical MARSGRAM atmosphere density predictions are shown on the lower right.
"Slalom Course" Maneuver Profiles

INM, RAAN, & ARCPER PROFILES

LIFT MODULATION VIA BANK ANGLE MOD.

D615-10026-2
According to MarsGRAM, the one sigma density variation about the mean is illustrated for day and night at the equator during spring equinox of the year 2016. The wider envelope for the equinox day profile is due to the diurnal bulge.
A set of synthetic-density wave equations are given on the following. These wave equations were used to determine aerocapture guidance sensitivities from density variations (worst case) that may be encountered during Mars Aerocapture.
Synthetic Density Wave Equations

- Horizontal sine-wave density scaling
  \[ DENS = DENS*\left[1.0 + 0.3*\text{SIN}\left((Y/W)*2\text{Pi}\right)\right] \]

- Horizontal sine-wave density and vertical density ratio scaling
  \[ DENS = DENS*\left[1.0 + 0.05*\text{SQRT}\left(DENS30/DENS\right)\text{SIN}\left((Y/W)*2\text{Pi}\right)\right] \]

- Horizontal sine-wave density and vertical density-ratio and sine-wave density scaling
  \[ DENS = DENS*\left[1.0 + 0.05*\text{SQRT}\left(DENS30/DENS\right)\text{SIN}\left((Y/W + H/LZ)*2\text{Pi}\right)\right] \]
Using the COSPAR low atmosphere with sine wave distribution of the density, calculations were made which illustrate that for sine wave lengths greater than 1000km the exit velocity errors are higher than for the lower wave lengths. The 30 - 40 km altitude region is by itself the most critical region.
EFFECTS OF HORIZONTAL DENSITY SINE-WAVE FORMS ON AEROCAPTURE EXIT VELOCITY

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REGIONS IN WHICH WAVE STRUCTURE IS ACTIVE:
- □ 0 TO 100 KM ALTITUDE
- + 70 TO 80 KM ALTITUDE
- ◇ 60 TO 70 KM ALTITUDE
- △ 50 TO 60 KM ALTITUDE
- ▼ 40 TO 50 KM ALTITUDE
- ▽ 30 TO 40 KM ALTITUDE
Guided trajectories were simulated for horizontal and vertical wave lengths. The horizontal wave length varied to 2000 km with vertical wave lengths of 5, 10, and 20 km. The larger vertical wave length of 20 km provides a more favorable atmospheric condition as the density variation in the vertical direction does not vary as much as the lower wave lengths in the critical region.
HORIZONTAL SINE-WAVE DENSITY AND VERTICAL DENSITY-RATIO AND SINE WAVE DENSITY SCALING
This page intentionally left blank
SCALING WITH SINE CURVE PHASE SHIFT

EXIT VELOCITY (KM/SEC)

TARGET VELOCITY

4.72

4.74

4.76

4.78

4.8

4.82

4.84

4.86

4.88

4.9

4.92

4.94

4.96

(Tens)

WAVE LENGTH (KM)

0

0.2

0.4

0.6

0.8

1

1.2

1.4

1.6

1.8

2

□ RANDOM

+ PI/2

○ 3PI/2
Aeroheating Methods and Assumptions

Stagnation Point Heating

- Radiative (Park Method)
  - Equilibrium (Stagnation Pressure > 0.1 atm)
  - Optically thick gas (absorptivity = 0.5)
  - Park Method reliable to within ±30%

- Convective
  - Modified Fay-Riddell
  - Fully Catalytic

Distributed Heating (Continuum flow)

- Radial Streamlines Assumed

- Radiation
  - Approximate shock shape used.
  - Averaged Normal Velocity Component is the used in the Park Method

- Convective (Boundary Layer Analysis Program)
  - Axisymmetric Analog
  - Pressure Distribution: Newtonian Impact Theory
  - Laminar Flow (Re transition = 2 \times 10^6)
MTV Aerobraking Constraints

The aerobraking constraints applied to the MTV configuration are summarized on the facing page. These constraints include center of mass location for trim at the desired L/D, keeping the MTV itself in the protected wake region.
0.5 L/D AEROBRAKE

Values for Shape Parameters

Semimajor Axis Ratio 2.00
Eccen. of Body of Revol. 1.60
Eccen. of Cutting Cyl. 0.40
Truncation/SMA ratio 0.00
Lip Radius/SMA ratio 0.05
Lip Taper Ratio 0.30

Normalized x length 4.07
Normalized Plan Area 11.8
Normalized Surface Area 15.4

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High L/D Aerobrake - Aerodynamic characteristics of the high L/D aerobrake, based on Modified Newtonian impact theory.
HIGH L/D AEROBRAKE

Values for Shape Parameters

Semimajor Axis Ratio 2.00
Eccen. of Body of Revol. 3.00
Eccen. of Cutting Cyl. 1.03
Truncation/SMA ratio 0.50
Lip Radius/SMA ratio 0.03
Lip Taper Ratio 0.30

Normalized x length 1.120
Normalized Plan Area 0.851
Normalized Surface Area 0.988

<table>
<thead>
<tr>
<th>Angle</th>
<th>CL</th>
<th>Cd</th>
<th>L/D</th>
<th>Moment Arm</th>
</tr>
</thead>
<tbody>
<tr>
<td>-10</td>
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<td>-0.1197</td>
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<tr>
<td>50</td>
<td>0.4974</td>
<td>0.4249</td>
<td>1.1706</td>
<td>0.4650</td>
</tr>
</tbody>
</table>
The stagnation point heating rates were calculated using the MARSIN code with a two dimensional trajectory and MarsGRAM atmosphere (high density). The convective heat transfer was calculated for a fully catalytic wall, the radiative heat transfer was calculated using the Tauber–Sutton method and equilibrium flow. The stagnation point radius of curvature was 13 m. The range of L/D varied from 1.0 to -1.0. For each condition a fixed L/D was used. The calculations illustrate that as L/D becomes negative the heating rate decreases.
Mars Aerocapture - MTV

Stagnation Point
Radius of 13 m

Ballistic Coefficient
394 Kg/m²

Radiation:
Tauber-Sutton Method

MarsGRAM
Atmosphere Hi Density
Mars Aerocapture Stagnation Point Heating - MTV

Aerocapture trajectory was computed with the MARSIN code. Initial conditions are given as follows:
- MTV hyperboloid aerobrake
- L/D = 0.5
- Flight averaged L/D = 0
- Approach C3 = 30 sq km/sq sec
- Entry altitude = 150 km
- Entry velocity = 7.4 km/sec
- Ballistic coefficient = 394 kg/sq m
- MarsGRAM high density (winter solstice) for 2016
Heating - MTV

Maximum Total Heat Flux at 120 sec

- Park Radiative
- Tauber-Sutton Radiative
- Fully Catalytic

Time, (sec)
Assumed Streamlines $S_1$, $S_2$, $S_3$

For an angle of attack of $20^\circ$

$L/D = 0.5$

Line of Symmetry

Stagnation Point

$R_a = 30\; \text{m}$

$R_a = 13\; \text{m}$
The following three charts show the pressure distribution along the fore, aft, and side streamlines based on a modified Newtonian theory, and unswept cylinder theory for the cylindrical lip.
Distribution

\[ P/P_{SP} \]

\[ s_1 (m) \]

\[ P_{SP} = \text{Stagnation Point Pressure} \]
Aft Centerline Pressure Distribution

\[ P_{SP} = \text{Stagnation Point Pressure} \]

---

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Side Streamline Pressure Distribution

\[ P/P_{SP} \]  

\[ S_3 \text{ (m)} \]

\[ P_{SP} = \text{Stagnation Point Pressure} \]
The following chart delineates the aero thermal conditions at the stagnation point for maximum heating along the aerocapture trajectory.
Conditions at Maximum Stagnation Point Heating

- Time - 120 sec
- Velocity - 6.714 km/sec
- Altitude - 41.2 km
- Density - $4.78 \times 10^{-7} \text{g/cm}^2$
- Radiative
  - Park - 116 w/cm²
  - Tauber-Sutton - 53 w/cm²
- Convective - 30 w/cm²
- Equilibrium wall temperature (emissivity = .8)
  - Park - 2300°k
  - Tauber-Sutton - 2070°k
- Maximum G-level = 3.0
For the hyperboloid aeroheader of approximately 30m length, the heating rate was calculated using the boundary layer analysis program (BLAP) for the convective heat transfer with the radiative heat transfer being calculated using the Park method and the Tauber-Sutton method. For the forward stream line heating, the radiative heat transfer using the Park method is approximately twice that of the Tauber-Sutton method. No turbulent transition was assumed for the calculation. The heating rate at the stagnation point is approximately 146 w/cm$^2$ using the Park method and approximately 80 w/cm$^2$ for the Tauber-Sutton method.
Aft Streamline Heating

ADVANCED CIVIL SPACE SYSTEMS

Heating Rates W/cm²

Total (Park)
Radiative (Park)
Total (Tauber-Sutton)
Radiative (Tauber-Sutton)
Convective (BLAP)

S₂ (m)
The Reynolds number was calculated for the aft stream line of approximately 30 meters. The Reynolds number was calculated based upon conditions behind the shock at the edge of the boundary layer utilizing the BLAP program. The calculations are for the highest heating rate at 120 sec. with a velocity of 6.7 km/sec at an altitude of 41.2 km. and a C3 of 30 km²/sec². The local Reynolds number at the rear of body is less than one million.
Local Reynolds Number Along Aft Streamline
Mars Transfer Vehicle
Incident Shock Layer Gas Radiation In the Base Region

Preliminary results of work performed under subcontract, by RemTech Inc. to ACSS is displayed in the following three charts. The study involved examining the base flow heating regime, which includes both convective and radiative heating for a C3 of 30 km²/sec² MTV aerocapture at Mars. The purpose was to determine the region of low heating behind the MTV and thus the protective cone for packaging the crew, habitat modules, and other cargo. The trajectory used for this analysis was provided by ACSS and is displayed on page 3-9, where the maximum stagnation point heating is 83 W/cm².

The graph below displays the base radiative heating rate as a function of time for the C3=30 trajectory. The equation shown on this chart gives the radiative heating rate to a surface in the base, for varying view angles. These radiative heating predictions are based on relationships derived and used for the AFE base flow heating regime. The maximum base flow heating occurred at 114 seconds with a maximum radiative rate of ~1.5 W/cm². This value is only 2.6% of the peak stagnation point value.
MARS TRANSFER VEHICLE
INCIDENT SHOCK LAYER GAS RADIATION IN THE BASE REGION

\[ \dot{q}_L = \dot{q}_{\text{peak}} (634 \cos \theta + 366) \]

Flow

Time seconds

Peak Incident Heat Rate WATT/CM²

0.0 0.2 0.4 0.6 0.8 1.0 1.2 1.4 1.6

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Mars Transfer Vehicle
Base Convective Heat Rates $T_w = 1367$

The graph below shows the convective heating rate within the base region along the C3=30 aerocapture trajectory. Maximum base convective heating occurred at 114 seconds, with a value of $\sim 1.7 \text{ W/cm}^2$. For the heating rate calculation, the assumed wall temperature ($T_w$) was 1367 K. Changing this $T_w$ value by $\pm 30\%$ resulted in only a $\sim 1\%$ change in heating rate and thus $T_w$ was found to be of small importance in computing the heating rate. Total heating rates for the base region are on the order of $3.2 \text{ W/cm}^2$, thus requiring a need for some TPS on surfaces within the base.
Shown below is a graphical depiction of the wake closure, and protective low heating region for the MTV aerobrake. This shear layer angle is based on calculations made for the expansion of the flow around the lip of the MTV aerobrake. Flow field properties around the lip were estimated using the BLIMPK program. The outer edge of the shear layer was calculated to be at 31° from the flow direction. The viscous region, estimated from experimental data, accounted for an additional 7° resulting in a total wake deflection angle of 38°. This preliminary estimate of the wake flow would impact the packaging of the aerobrake contents.
The following approximate temperature contours are based on the
BLAP convective and Tauber_Sutton radiative streamline heating dis-
tribution.
Tauber-Sutton Radiation

**Trajectory**
- C3 = 30 km^2/sec^2
- Flight Averaged Lift = 0
- L/D = 0.5
- Ballistic Coefficient = 39.4 g/cm^2
- Max G-level = 3.0

**Conditions at Max Heat Rate**
- Time = 120 sec
- Velocity = 6.714 km/sec
- Altitude = 41.2 km
- Density = 4.78x10^-7 g/cm^2
- Total Heat Rate = 83 w/cm^2
- Stagnation Temp. = 2069 K

**Total Heat Load**
- Q = 5,880 j/cm^2

**Emissivity = 0.8**
Preliminary Results

Stagnation Point Heating (Park Method)

- Radiative heating for $30 < C_3 < 50 \text{ km}^2/\text{sec}^2$
  $\geq 80\%$ of total (Park)

$\geq 65\%$ of total (Tauber-Sutton)

- Stagnation temperatures for $C_3 \geq 30 \text{ km}^2/\text{sec}^2$
  exceed $2000^\circ \text{k}$

<table>
<thead>
<tr>
<th>$C_3$</th>
<th>Park</th>
<th>Tauber-Sutton</th>
</tr>
</thead>
<tbody>
<tr>
<td>30</td>
<td>146.</td>
<td>83.</td>
</tr>
<tr>
<td>40</td>
<td>299.</td>
<td>170.</td>
</tr>
<tr>
<td>50</td>
<td>481.</td>
<td>274.</td>
</tr>
<tr>
<td>1993 technology</td>
<td>68.</td>
<td>2068</td>
</tr>
</tbody>
</table>
The stagnation point heating rates were calculated using the MARSRT* code with a two dimensional trajectory. The convective heat transfer was calculated for a fully catalytic wall; the radiative heat transfer was calculated using the Park method and equilibrium flow. The stagnation point radius of curvature was 13 m. The range of L/D varied from 1.0 to -1.0. For each condition a fixed L/D was used. The calculations illustrate that as L/D becomes negative, the heating rate decreases.

*MARSRT - (mars return) an aerocapture at Earth heating code
**Earth Aerocapture - MTV**

**Convective Rate $W/cm^2$**
- 6-g limit
- $C_3 = 30$
- $C_3 = 40$
- $C_3 = 50$

**Radiative Rate $W/cm^2$**
- 6-g limit
- $C_3 = 30$
- $C_3 = 40$
- $C_3 = 50$

**Total Rate $W/cm^2$**
- 6-g limit
- $C_3 = 30$
- $C_3 = 40$
- $C_3 = 50$

**Stagnation Point**
- Nose Radius 13 m

**Ballistic Coefficient 225 kg/m$^2$**

**Radiative Heating**
- Park Method
For an average L/D of 0.5 using the same relationships as previously, the Earth aerocapture total heating rate is 172 w/cm² using the Park method. The peak convective heating rate was about 70 w/cm² calculated by the BLAP code. The earth aerocapture stagnation heating rate is higher than the stagnation point heating rate for Mars which was calculated to be 146 w/cm².
Two structural designs, spar and truss, were used for the hyperboloid aerobrake. The aerobrake has a length of approximately 31 meters, width of 28.2 meters and height of 6.5 meters. In both cases the aerobrake accommodates an engine hatch. In the case of the truss configuration, the tetrahedral truss respective points were projected on to the aerobrake surface to accommodate the required curvature. The spar configuration used a carbon magnesium metal matrix and the truss configuration used graphite epoxy. In both cases an aluminum honeycomb core was employed with a titanium face sheet. The Mars excursion vehicle had a payload of 81 metric tons and the Mars transfer vehicle the payload was 153 metric tons.
Design Assumptions:

- Constant spar cross-sections, curved profiles
- C/Mg metal matrix spars (density 1830 kg/cu. m.)
- Payload: Mars Excursion Vehicle, 81MT
- 6g maximum acceleration
- 8 payload attach points (4 frame and 4 landing leg points)
- Relative wind angle = 20 degrees
- Variable pressure distribution, range 1.5psi to 3psi
- Structure temperature = 394K (250F)
- Secondary spar pattern to be triangular for greater shear resistance
## Aerobrake Structural Design
81 mt payload, MEV

<table>
<thead>
<tr>
<th>19.5 inch spar depths:</th>
<th>105 ksi spar strength</th>
<th>200 ksi spar strength</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Primary spar weight:</strong></td>
<td>5,390 kg (11,859 lb)</td>
<td>2,571 kg (6,052 lb)</td>
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<tr>
<td><strong>Secondary spar wt:</strong></td>
<td>3,827 kg (8,420 lb)</td>
<td>2,975 kg (6,546 lb)</td>
</tr>
<tr>
<td><strong>Honeycomb weight:</strong></td>
<td>6,758 kg (14,868 lb)</td>
<td>6,758 kg (14,868 lb)</td>
</tr>
<tr>
<td><strong>TPS weight:</strong></td>
<td>3,300 kg (7,260 lb)</td>
<td>3,300 kg (7,260 lb)</td>
</tr>
<tr>
<td><strong>Total aerobrake weight:</strong></td>
<td>19,275 kg (42,407 lb)</td>
<td>15,784 kg (34,726 lb)</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>22.5 inch spar depth:</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Primary spar weight:</strong></td>
<td>4,989 kg (10,978 lb)</td>
<td>2,484 kg (5,465 lb)</td>
</tr>
<tr>
<td><strong>Secondary spar wt:</strong></td>
<td>3,809 kg (8,379 lb)</td>
<td>2,596 kg (5,711 lb)</td>
</tr>
<tr>
<td><strong>Honeycomb weight:</strong></td>
<td>6,758 kg (14,868 lb)</td>
<td>6,758 kg (14,868 lb)</td>
</tr>
<tr>
<td><strong>TPS weight:</strong></td>
<td>3,300 kg (7,260 lb)</td>
<td>3,300 kg (7,260 lb)</td>
</tr>
<tr>
<td><strong>Total aerobrake weight:</strong></td>
<td>18,856 kg (41,483 lb)</td>
<td>15,138 kg (34,726 lb)</td>
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</tbody>
</table>

Note: 200 ksi option may require additional material technology development efforts.
Many state-of-the-art advances are needed to support on-orbit assembly and checkout

- Robotics
- Non-destructive inspection
- "Smart" structure
- Joint closure

Advanced Thermal Protection System (TPS) materials are essential for C3 > 30; temperatures > 1800°C will probably require a mass penalty
Aeroheating Principal Findings

The findings from our current analysis are that Mars capture aeroheating is a significant problem, unless correlations that predict less radiative heating can be verified. Several work-arounds are noted on the chart. In the next three months, we will be working with alternative radiation correlations and exploring the efficacy of these work-arounds.
Aeroheating Principal Findings

• Mars Aerocapture Radiation Problems

- For C3's from 30-50 km²/sec², Radiative flux is 80-90% of total heat flux.

• Stagnation Temperatures for C3's ≥ 30 are above near term reradiative technologies.

<table>
<thead>
<tr>
<th>C3</th>
<th>Q(w/cm²)</th>
<th>T °K</th>
</tr>
</thead>
<tbody>
<tr>
<td>30</td>
<td>146.</td>
<td>2383</td>
</tr>
<tr>
<td>40</td>
<td>299.</td>
<td>2850</td>
</tr>
<tr>
<td>50</td>
<td>481.</td>
<td>3210</td>
</tr>
</tbody>
</table>

Note: 1993 reradiative technology = 68 w/cm² or = 1968 °K

• Options

- Use of ablators for Mars Aerocapture.
- Improve reradiative materials.
- Limit the Missions to lower approach C3's.
- Modify or Change the Aerobrake shapes.
- Optimize Trajectories for mininal aeroheating (Down-Lift).

• Needs

- Improved Analysis for Non-Equilibrium Radiation in CO₂ Atmosphere
- Engineering Methods for treating Non-Axisymmetric blunt body flows.
Importance of Landing Site Analysis

The reasons for performing landing site analyses are indicated on the facing page. Landing site access will be the requirements driver for Mars Excursion Vehicle aerobrake L/D and descent profile design.
Importance of Landing Site Analysis

- Landing site location determines the L/D requirements to get from orbit to the site

- L/D requirements determine the configuration of the Mars Excursion Vehicle Aerobrake

- The configuration of the aerobrake determines the load points, vehicle stress and available wake cone area to place the lander vehicle inside of

- The wake cone, stress points, and load points determine the configuration of the lander

- The size and shape of the aerobrake can determine the amount of packaging required for ETO launch and the number of launches required if it is assembled in space
Preliminary Mars Landing Sites Between ± 20° Latitude

The next three pages show a sampling of landing sites of scientific interest in the ± 20° latitude band on Mars. Altitudes are also shown, since altitude has a strong effect on landing delta V. We are presently designing for access to any site within this latitude range, at altitudes up to 5 km. with the COSPAR low-density atmosphere. This will permit landings up to 8 - 9 km altitude with typical atmosphere densities.
### Preliminary Mars Landing Sites Between +/- 20° Latitude

<table>
<thead>
<tr>
<th>Place</th>
<th>Planet coordinates</th>
<th>Martian altitude</th>
<th>Areas of Interest (accessible by rover, 1000km out from landing)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tharsis Montes</td>
<td>5°N 100°</td>
<td>9 km</td>
<td>Ascræus and Pavonis Mons, rill formations, Tharsis Tholus, unnamed crater</td>
</tr>
<tr>
<td></td>
<td>10°N 82°</td>
<td>3-2 km</td>
<td>Tharsis Tholus, Echus Chasma, Fesenkov Crater, head of Kasei Vallis, Lunae Planum (colored soil)</td>
</tr>
<tr>
<td></td>
<td>-10° S 137°</td>
<td>4 km</td>
<td>Mangala Vallis, Memnonia and Sirenum Fossae, edge of Tharsis Montes shield, Aganippe Fossa, Arsia Mons Colored sands</td>
</tr>
<tr>
<td></td>
<td>-15° S 115°</td>
<td>8-9 km</td>
<td>Arsia Mons, Noctis Labyrinthus, Syria Planum, Claritas Fossae, crater area</td>
</tr>
<tr>
<td>Sinai Planum</td>
<td>-18°S 76°</td>
<td>1-8 km</td>
<td>Melas Chasma of Valles Marineras (possible access to Valles Marineras floor); Felis, Melas and Solis Dorsii, crater (unnamed)</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>with rills/flows, Lassell Crater, Coprates Catena</td>
</tr>
</tbody>
</table>
### Preliminary Mars Landing Sites Between +/- 20° Latitude

<table>
<thead>
<tr>
<th>Place</th>
<th>Planet Coordinates</th>
<th>Martian Altitude</th>
</tr>
</thead>
<tbody>
<tr>
<td>South of Eos Chasma</td>
<td>-19°S 49°</td>
<td>3 km</td>
</tr>
<tr>
<td>Hesperia Planum</td>
<td>-16°S 253°</td>
<td>4 km</td>
</tr>
<tr>
<td>Elysium-Amazonis</td>
<td>0° 180°</td>
<td>0 km</td>
</tr>
</tbody>
</table>

**Areas of Interest (accessible by rover, 1000km out from landing):**

- Eos Chasma (part of Valles Marineris, with possible access to the valley floor, accessible places in the valley floor -1 and -2 km)
- Lassell and Ritchey Craters, Helis Dorsa, crater field some with flow fields, Holden Crater
- Tyrrhena Patera (massive flow field from a single source), crater fields, surface cracks and fissures, Terra Tyrrhena area, small mounts
- Pettit Crater, Nicholson Crater, surface cracks, Orcus Patera, Cerberus Rupes, colored soils, old craters, Apollinaris Patera, Gusev Crater and flow field, edge of Elysium flow shield, Medusae Fossae, "new" craters in the Elysium flow shield
## Preliminary Mars Landing Sites Between +/- 20° Latitude

### Place

<table>
<thead>
<tr>
<th>Place</th>
<th>Planet coordinates</th>
<th>Martian altitude</th>
<th>Areas of Interest (accessible by rover, 1000km out from landing)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Amazonis Planitia</td>
<td>15°N 155°</td>
<td>0-3 km</td>
<td>flow area around Olympus Mons, edge of Gordii Dorsum and Eumenides Dorsum formations, crater area east of Pettit Crater</td>
</tr>
<tr>
<td>Chryse Planitia</td>
<td>18° N 45°</td>
<td>0(-1) km</td>
<td>Chryse depression (-3 km), end of Kasei Vallis, Sharonov Crater, Lunae Planum; Nanedi, Shalbatanu, Simud, and Tiu Valles, end of Ares Vallis, craters, colored sands</td>
</tr>
<tr>
<td>Place</td>
<td>Planet coordinates</td>
<td>Martian altitude</td>
<td>Areas of Interest (accessible by rover, 1000km out from landing)</td>
</tr>
<tr>
<td>-----------------------</td>
<td>--------------------</td>
<td>------------------</td>
<td>--------------------------------------------------------------------------------------------------------------------------</td>
</tr>
<tr>
<td>North of Ganges</td>
<td>-2° S, 68°</td>
<td>1 km</td>
<td>Ophir Chasma (part of Valles Marineras, with possible access to the valley floor), Hebes Chasma, Echus Chasma, Juventae Chasma, Ophir Planum, Lunae Planum, crater field, colored soil</td>
</tr>
<tr>
<td>Catena</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Elysium Planitia</td>
<td>19°N, 226°</td>
<td>0 km</td>
<td>Hephaestus Fossae, Elysium Fossae, Elysium Mons, Albor Tholus, Eddie Crater with interior formation, colored sands</td>
</tr>
<tr>
<td></td>
<td>19°N, 197.5°</td>
<td>2-3 km</td>
<td>Elysium Mons, Elysium Fossae, Ocrus Patera, Cerberus Rupes, Lockyer Crater, Phlegra Montes, colored sands, craters, old and &quot;new&quot;</td>
</tr>
</tbody>
</table>
Mars Descent Analysis Findings

- Wide variation in vehicle crossrange achievable with $0.5 < \text{L/D} < 1.0$

- Latitudinal displacement of $> 20$ deg is achievable for a wide range of entry conditions, for $\text{L/D} > 0.9$

- Increasing descent vehicle L/D provides diminishing Delta V requirements

- Landing altitudes of $5$ km to $10$ km above Mars reference are achievable with present crossrange requirements, for $\text{L/D} > 0.95$

- High L/D shape will be a flatter shape then previously investigated with a partial top shield to control wake impingement

- A flare aeromaneuver will be performed at the end of the landing sequence (before vehicle landing for the reusable aerobrake and aerobrake drop for the non-reusable Level II scenario). This flare will be controlled by:
  - For the non-reusable aerobrake, a large flap (.25 of aeroshell area) will separated prior to aerobrake drop.
  - For the reusable aerobrake, the center of gravity control, and therefore the flare maneuver, will be managed by pumping LOX to and from main and auxiliary tanks
  - The reusable aerobrake will also have an articulated flap with a maximum area of 10% of the aerobrake
The aerobrake configuration utilized in this analysis is the same as previous. With an L/D of 1.0 the flight time is 2 times greater than for an L/D = 0.5, thereby resulting in range increase of 50%.
Landing Analysis
Range Effect of L/D

Entry Mass
Isp
Ref Area

81 mt
470 sec
471 m²

L/D = 1/2

L/D = 1.0

Time in Seconds

Altitude in km

/21895/AEROBRAKE STUDY/DISK 3/5/66 0/10:00 A
/STCAEM/Aev/31May90
The following plots were generated the using OTIS (Optimal Trajectories by Implicit Simulation) program. The plots show the flight path of a vehicle that will give the maximum crossrange and they show the crossrange. The plots were done for a vehicle flying at a lift to drag ratio of one and a lift to drag ratio of one-half.
Lift to Drag Ratio = 0.5
The hyperbolic shaped vehicle analysis was carried out for a range of L/D from 0.65 to 1.0. The reference area of the aerobrake was 471 square meters with a total weight of 81 metric tons. The aerobrake weight itself was 9 metric tons. For the analysis, the drag coefficient was held constant at 0.75 and the angle of attack was held at 20°. The calculation was carried out for the cool low density COSPAR atmosphere. For an L/D = 1.0 a change in latitude of 30° is achievable. For the same conditions, a crossrange of 1000 km may be obtained.
Landing Analysis
Delta Latitude

Engine Start Altitude = ESA
Orbital Inclination = I
Mars Landing Simulation with Aerodynamic Flare

Shown on this chart are results of an un-optimized, but typical, aerodynamic and propulsive descent. Mars entry occurs at a 90° roll angle to obtain maximum cross-range. As the vehicle slows to below circular velocity, roll-out in two steps maintains roughly level flight. Most of the descent is flown at \( L/D = 1 \). Prior to engine start, the \( L/D \) is briefly increased (drag decreased) to increase speed. Then the vehicle is pitched to maximum lift coefficient at \( L/D \) about 0.5. This causes an aerodynamic flare, decreasing speed and increasing path angle. The result is a significant decrease in rocket thrust and delta V for landing.

The importance of this is that it generates a requirement for pitch control, a requirement not present for aerocapture. The combination of high \( L/D \) and pitch control will lead to selection of an aerobrake shape much different from the MTV aerocapture case.
Mars Landing Simulation With Aerodynamic Flare

- COSPAR low-density atmosphere
- Entry mass 811
- Thrust 80k
- Max L/D 1.1
- Ref Area 750m²

---

D615-10026-2
For a range of L/D from 0.65 to 1.0, the ideal delta velocity range is from 1400 m/s to 1200 m/s for landing at a 5 km altitude. In the case of an aeroflare with two different reference areas of 470 and 750 m² with a cutoff velocity 0.02 the ideal delta velocity is approximately 900 m per second. This results in a delta velocity reduction due to using the aeroflare from 200-300 m/s for L/D = 1.0.
Ideal Delta Velocity

Reference Area = 471 sq. m
Thrust = 420 kn
ESA = 11.4 km
Aerobrake Drop = 6.79 km
Cutoff Velocity = 0.1 km/s
Ideal Del Velocity = 529 m/s

Reference = 750 sq. m
Thrust = 420 kn
ESA = 11 km
Aerobrake Drop = 8.52
Cutoff Velocity = 0.02
Ideal Del Velocity = 918.5
Levied Requirements
Reference Cryogenics/Aerobrake (Cryo/Aerobrake) - System Requirements

During the course of the Space Transfer Concepts and Analysis for Exploration Missions contract (STCAEM), Boeing's Advanced Civil Space Systems group (ACSS) has conducted regular review meetings in order to define and derive requirements, conditions and assumptions for systems currently being developed.

As system definition and development progresses, technical experts provide documentation and rationale for requirements that have been derived. This real-time capturing prevents requirements and their associated rationale from being lost or neglected. For example, a vehicle configurator may see the need for providing a minimum passage dimension for vehicle egress or ingress. This requirement would then be captured at an early development stage and would provide a history for the decision. This seemingly simple requirement may have large impacts on the design down the road and its traceability is important.

Derived requirements and rationale are later transferred to the Madison Research Corporation (MRC) where they are then entered into the system data base which has been developed for ACSS using ACIUS's 4th Dimension® software. The data base allows for easy access and traceability of requirements.

The charts that are contained within this document represent two collated copies of principal requirements and assumptions for February 2, and May 30, 1990. The systems defined include: (1) the Mars Transfer Vehicle (MTV), (2) Mars Excursion Vehicle (MEV), (3) Trans-Mars Injection Stage (TMIS), and the Earth Crew Capture Vehicle (ECCV). Each system is then broken down into subsystem headings of: (1) design integration, (2) guidance, navigation and control (GN&C), (3) electrical power, (4) man systems, (5) structure and mechanisms, (6) propulsion, (7) ECLSS, (8) and command and data handling (C&DH). The initials of each of the technical experts responsible for developing the supporting rationale for each of the requirements is indicated parenthetically next to each entry.

Although the majority of the derived requirements listed are directly applicable to all vehicles such as those powered by Nuclear Electric propulsion (NEP), Nuclear Thermal Rockets (NTR), Solar Electric propulsion (SEP) and reference Cryo, there are some that are not. Those requirements that are only directly applicable to a specific vehicle type are indicated within the entry. The italicized entries indicate a modification to an original requirement prior to the second revision of May 30, 1990.

Defining and re-examination of derived requirements will continue through the current contract.
General Requirements

- First Mars landing in 2016
- Cargo flight and second manned landing in 2018
- Vehicles sized to meet given mission phase ΔV budget and durations
- Manned flights to deliver crew and 25t payload to surface
- Crew and 1t payload returned to LEO for each manned flight
- NASA STD 3000 applicable
- Quarantine and medical provisions provided
- Factors of safety set for metallic/nonmetallic/pressure structures
- Failure tolerance/ maintainability requirements identified
Mars Transfer Vehicle (MTV)
- Aerobraking to be used at Mars arrival \( (V_e \leq 9500 \text{ m/sec}) \)
- Aerobraking to be used at Earth return \( (V_e \leq 12500 \text{ m/sec}) \)
- Provide direct entry capability at Earth return \( (V_e \leq 14000 \text{ m/sec}) \)
- Piloted MTV to be reusable without major maintenance for 5 missions
- EVA capability provided
- Zero-G transit
- In transit science performed outbound and inbound

Mars Excursion Vehicle
- Expendable Vehicle
- Chemical propulsion (LOX/LH2)
Derived Requirements
MTV Derived Requirements

- **Design Integration**
  - Two (2) communications satellites deployed in Mars orbit with total mass = 3000kg (GW)
  - Crew module must accommodate alternative advanced propulsion options (BD)

- **GN&C**
  - Capture trajectory entry interface for aerocapture not to exceed 6'g' limit and to preclude an uncontrolled skip-out (PB)

- **Electrical Power**
  - Solar power to be used for transfer phase, batteries to be utilized for sun occultation time while in Mars orbit (BC)

- **Man Systems**
  - Added protection to crew from Solar Proton Events (SPE) will incorporate use of a "storm shelter". (MA)
  - Consumables stored will suffice for crew residence time from 443-1018 days (includes abort), assumes 100% ECLSS closure of water and oxygen, 0% closure on food and .25 kg leakage per day (PB)
  - Two (2) astronauts able to pass through major circulation paths while wearing EVA suits. (SC)
  - Crew quarters shall provide sufficient volume for casual conversation between at least two (2) crew members (SC)
MTV Derived Requirements (continued)

- Man Systems (continued)
  - Crew visibility during all maneuvers (docking/rendezvous) (SC)
  - Crowd module to accommodate 0g and induced g environments (SC)
  - There shall be 2 means of egress from each module for emergency escape (SC)

- Structure and Mechanisms
  - Airborne support equipment for aerobrake shall be 20% of aerobrake mass (PB)
MEV Derived Requirements

- **Design Integration**
  - Provide 15% of active weight for spares (JM)
  - MAV must be able to abort-to-orbit during descent phase (PB)
  - Twenty-five (25) ton down payload on manned vehicles (BS)
  - Protective covers provided for all mission critical systems (BS)

- **GN&C**
  - L/D range from 0.5 to 1.0 (GW)
  - Deorbit from 1 sol x 250 km periapsis orbit (GW)
  - Currently, cross range = ± 500km (GW)
  - Engine start before aerobrake drop (GW)
  - Approach path angle = 15° (GW)
  - Capture trajectory entry interface for MEV aerocapture at Mars not to exceed 6'g' limit on crew members and equipment and to preclude an uncontrolled skipout of the Mars atmosphere (PB)
  - Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1km cep and with beacon assuming 30m cep (PB)
  - Autonomous aerocapture capability at Mars, ~one (1) day before MTV (BS)
  - Aerobrake jettisoned in controlled manner during powered descent phase (BS)
MEV Derived Requirements

(Continued)

- **Propulsion**
  - Pre-descent checkout of engines to be provided (checkout extent TBD) (BD)
  - One (1) meter clearance established between engine bells and surface (SC)

- **Electrical Power**
  - Solar arrays to supply power to MEV following separation from MTV for fifty (50) day approach to Mars (BC)
  - Power for 50 day approach sequence to Mars shall be provided by solar arrays separate from the full MTV configuration. Arrays to be retracted 12 hours prior to Mars encounter, power shall be provided by batteries or other internal source (BC)

- **ECLSS**
  - Capability of two (2) crew cab represses (BD)

- **Man Systems**
  - Consumables will suffice for a crew residence time of 30 - 600 days dependent on mission stay time and abort scenarios, assumes 100% ECLSS closure of water and oxygen, 0% closure of the food and .15 kg leakage per day (PB)
  - The maximum surface stay time is 600 days (PH)
- Structure and Mechanisms
  - Shall be at least two (2) functionally independently pressurized areas for emergency conditions
    The shall be two (2) EVA suits stored in these areas (PB)
  - Establish 30cm clearance between all elements to allow for movement during high-stress maneuvers (SC)
  - Crew cab to have SSF diameter (4.4m), width (1.4m), and penetrations and attachments occur at rings. (SC)
  - Surface hab system to be: removable later by surface construction transport vehicle and protected from damage by MAV blast during ascent start (BS)
MTV - TMIS Derived Requirements

- **Design Integration**
  - Flexible to support reference missions (interconnect design to support reference mission requirements (GW))
  - Fully modularized to utilize ETO capacity, the amount of modularization shall be a function of the ETO vehicle chosen (PB)
  - Assembly to be accomplished on-orbit, remotely and robotically (BS)

- **Propulsion**
  - Reference vehicle is launched "wet" with top-off (dry/wet issue to be traded) (JM)

- **Structure and Mechanisms**
  - Thrust structure - tanks - intertanks used as primary structure (GW)
  - The airborne support equipment mass for launch to Earth is assumed to be 7% for all hardware sets (PB)
Mars Transfer System Derived Requirements

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide (BS)
  - Equipment design life must account for mission duration plus one year (BS)
  - All components designed for 5 missions with refurbishment (except aerobrake) (BS)
  - Design for range of crew sizes, from 4 to 12 (BS)
  - L/D range from 0.5 to 1.0 for aerobrake vehicles at Mars (BS)

- **GN&C**
  - 8500 m/s maximum entry velocity at Mars (GW)
  - 100 m/s error-correction (post aerocapture) (GW)

- **Propulsion**
  - Engine out capabilities in all mission phases (BD)
  - Engine must continuously track C.G. of vehicle from beginning to end of all burns (BD)
  - Maximum gimbal angle of engines TBD (BD)

- **Man Systems**
  - Solar Proton Event (SPE) protection to be provided (MA)
  - Allow for direct viewing of all docking, berthing and landing procedures (SC)
• **Structure and Mechanisms**
- All critical function lines and redundant systems shall run non-parallel (PB)
- All systems shall function up to 2 years in a dormant state and having been subjected to the harsh space environment (PB)
- The airborne support equipment mass for launch to Earth orbit shall be assumed to be 15% for all hardware except the aerobrake (PB)
- Airborne support equipment mass assumption for the aerobrake shall be 20% of the aerobrake mass (PB)
- Aerobrake will be launched to Earth orbit in sections for on-orbit assembly as the reference case (PB)
- MTV and MEV aerobrakes have common layout of attach points (BS)
- Vehicle elements will have removable debris shield panel cladding for protection during LEO operations. These panels will be removed and saved in LEO to be used for the next mission-opportunity. The panels will not add to the LEO debris environment (BS)
- Mission vehicles will carry a robotic manipulation capability to inspect and maintain all exterior areas and systems (BS)
- Structure optimized to minimize weight, operations, complexity and development effort (BS)
- Greater than 30cm separation between all major vehicle exterior systems (i.e., tanks, modules) (BS)

• **C&DH**
- Connectability between links maintained 90% of the time. Availability when scheduled - 98% connectability (PH)
MTV - ECCV Derived Requirements

- **GN&C**
  - Capture trajectory entry interface for ECCV aerocapture or aeroentry into Earth atmosphere not to exceed 6\(^g\) limit on crew and personnel, and to preclude an uncontrolled skip out of Earth atmosphere (PB)
  - L/D = 0.25 (MF)

- **Structure and Mechanisms**
  - Interior materials must conform to NASA standards for outgassing, fire hazards, etc. (SC)
MTV - TMIS Derived Requirements

- **Design Integration**
  - Assembly to be minimized to extent practical (KS)

- **Propulsion**
  - Passive thermal control system including zero-'g' thermodynamic vent system coupled to multiple vapor cooled shields (JM)
  - TMIS insulating system is a continuously purged MLI over foam design optimized for minimum ground-hold, launch, and orbital boiloff. Includes vapor cooled shield (coupled to TVS) outside of foam (JM)
  - TMIS tanks launched late in assembly sequence to minimize orbital stay time before TMI burn (6 months). (JM)
  - MTV tank insulation system is thick (2-4") MLI blankets. Multiple vapor cooled shields placed at optimum points in the MLI. (JM)

- **Structure and Mechanisms**
  - Thrust structure - tanks - intertanks used as primary structure for cryo/aerobrake only (GW)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics
Mars Transfer System Derived Requirements

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide. The total wake closure angle is centered on the velocity vector. (BS)

- **GN&C**
  - 200 m/s error correction (post aerocapture) (GW)

- **Propulsion**
  - *Engine out capabilities in all mission phases. NTR engine out capabilities TBD* (BD)
  - All passive cryogenic thermal control system.
  - No. MTV-TMIS fluid transfer before Earth departure. (MEV tanks refrigerated or filled after MOI)

- **Structure and Mechanisms**
  - Aerobrake externally mounted to vehicle for launch to Earth orbit ("Ninja Turtle" concept) (PB)

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*Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics*
MEV Derived Requirements

- **Design Integration**
  - Down payload on manned vehicles
    - ~ 25 mt down payload for reference MEV (includes habitat module) (BD)
    - ~ 0.7 mt down payload for the 'Mini-MEV' (crew habitat is provided by the ascent/descent cab) (BD)

- **GN&C**
  - *Currently, cross range = ± 1000 km for high L/D aerobrake (GW)*
  - *Landing approach path angle = 15° (GW)*
  - *Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1 km CEP and with beacon assuming 30 m CEP (PB)*

- **Propulsion**
  - Engine out capabilities for ascent/descent stages (BD)
  - Passive cryogenic storage system: MLI with vapor cooled shields (JM)
  - Gravity field environment eliminates need for zero-'g' acquisition and venting. (JM)
  - Vacuum jacketed ascent tanks for Mars boiloff reduction. (JM)
  - MEV propellant transferred from MTV prior to descent. (JM)

- **Electrical Power**
  - *Solar arrays to supply power following separation from MTV for ~ 50 day approach to Mars. Arrays to be retracted TBD hrs. prior to Mars descent (cryolaerobrake). (BC)*
  - Batteries or fuel cells to provide power for ascent and descent phases. (BC)

**Note:** Changes to existing derived requirements dated 02 February 1990 are shown here in italics.
MTV Derived Requirements

- **GN&C**
  - Capture trajectory entry interface for aerocapture options not to exceed 6\(\cdot\)g' limit and to preclude an uncontrolled skip-out (MC)
  - Aerocapture exit errors not to exceed 0.25\(^\circ\) inclination, RAAN, and ARCP, and a 0.1 hr period (MC)
  - GN&C requirement for advanced propulsion TBD: (MC)
    - NTR - capture into planned orbit ± TBD
    - EP (electric propulsion options) - TBD

- **Electrical Power**
  - Solar power to be used for transfer phase, batteries or fuel cells to be utilized for sun occultation time while in Mars orbit except for NEP. (BC)
  - NEP power derived from existing power system with a backup energy supply via fuel cells (BC)

- **Man Systems**
  - Volume per crew guidelines extrapolated from historical data (SC)
    - Transfer hab = 112 m\(^3\)/crew
  - Two independent pressurized volumes for safety (SC)
  - Gravity condition emphasized to accommodate 0\(-\cdot\)g' and 1\(-\cdot\)g' and for surface commonality (SC)
  - 2.3 m standard ceiling height for psychological and locomotion (SC)

- **Structure and Mechanisms**
  - All penetrations occur in barrel section to minimize mass. (SC)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics.
Guidelines and Assumptions
Mars Initiative - Assumptions

- Multi-impulse TMI and TEI is permitted, (engine restart)
  (e.g., three-burn departures acceptable for TMI to ease launch declination window
  problems [Level II])

- Cryogenic propulsion for Earth/Mars departures and Mars descent (cryogenic/aerobrake for
  Earth and Mars are selected as reference)

- Proven cryogenic storage technologies will be used

- Advanced propulsion technology options include NTR, SEP, NEP, and GCR

- MTV expendable on "difficult" opposition missions; return to Earth via ECCV

- TMIS expendable for reference system

- 100 ton cargo requirement (cargo mission) met by two (2) standard MEV's without ascent
  stages

- Maximum size surface payloads on piloted MEV: 6 m diameter and 13 m length
Contingency, Flexibility and Reserves

**Flight Performance Requirements and Reserves**

- 2% ΔV for Space Transfer Vehicles
- Add 2% for performance requirements uncertainties in selected instances
- Compute finite burn losses and add to impulsive requirements.
- Include delta V requirements for launch windows from LEO.

**Dry Mass Contingency Allowances**

- None for existing hardware
- None for consumables and impulse propellant
  - Consumables requirements shall include needed mission flexibility allowances.
  - Propellant reserves generated by flight performance reserves.
  - Use 2% of tank capacity for liquid and vapor unusable propellant; counts as inert mass.

- 5% on slightly modified hardware
- 15% on new design/known technology
- 15%-25% on new design/new technology, complex design, and poorly-understood requirements.

**Payloads include flight support equipment**

**Manager's Reserve Policy, e.g. between launch vehicle capability and manifesting, TBD.**
III. Operating Modes and Options
Reference
Cryo/Aerobrake - Operating Modes and Options Reference

This section contains the following:
- Operations Outline
- Operations Task Flow Description
- Operations Assumptions
- Operational Task Flow

In order to evaluate the difficulty of the mission operations a top level view of the necessary sequence of events was generated. Only the areas of on-orbit assembly and ground support were delved into to any depth. These are discussed in the support section of this document. The path itself is shown in this section and includes options at assembly (on or off Space Station), in transit outbound or inbound (with or without Venus flyby, Deep Space Burn and coast correction any combination of which may be used) and on return (depending on how much of the vehicle is recovered and where it is recovered at).

The Cryo/Aerobrake vehicle will operate out of the LEO Space Station orbit. The completed vehicle will leave from a position co-orbiting with the Space Station and do one to three burns to attain the Declination Launch Asymptote (DLA) required for Mars transfer. The Trans-Mars - Injection stage is dropped after the last burn and the transit configuration established. This, at present, is the zero-gravity transit configuration, but an artificial gravity configuration would be established at this point in flight. For any swingby, Deep Space Burn or coast correction maneuver, the artificial gravity configuration must be despun, reconfigured to the zero-g conditions and reconfigured to the artificial gravity conditions after the maneuver has been performed.

The on-line self-check capability of the systems and subsystems will be used throughout the mission to monitor the vehicle health and indicate preventative maintenance. Due to the length of the mission (1-3 years) the vehicle must be self sufficient and capable of maintenance and repair with a limited crew (4-7 people). The length of mission time and the distance will impose limits on the communications and control of the vehicle that can be done by ground operations; the crew are on their own resources.

About 50 days prior to Mars entry the Mars Excursion Vehicle (MEV) and the Mars Transit Vehicle (MTV) will separate, with the MEV operating autonomously and entering first as a pathfinder, the two vehicle sections will aerocapture and rendezvous in orbit. If anything happens to the MEV in capture, the MTV with crew, will abort and return to Earth. After the vehicle sections are docked and the final site selection has been made, the MTV will be set to operate autonomously, the crew will transfer to the MEV, demate the MTV and MEV, perform the on orbit checkout and descend to the surface. The MEV will have the capability to perform a descent abort with the ascent section in the event of an emergency to obtain orbit. From there, a rendezvous and docking maneuver with the MTV will be done for crew transfer and Earth return.

Once on the surface, the MEV establishes contact with both the automated MTV and Earth, then proceeds to carry out the surface mission. When the surface mission is complete, the ascent section liftoff leaving the descent section of the lander and surface habitat behind. The ascent section attains orbit and docks with the MTV, the crew transfers with the return samples and all extraneous mass is jettisoned prior to the Trans-Earth -Injection Burn.

The inbound return transit proceeds like the outbound leg, with options in Venus swingby, coast maneuvers and transit flight configuration. On Earth return, the baseline option is to have the crew and samples transfer to the Earth Crew Capsule Vehicle (ECCV) several days before Earth entry takes place, disengage from the MTV and return to Earth on a direct entry course in the style of the
Apollo crew capsule. Alternate capture scenarios involve capturing the ECCV into a Space Station access orbit and crew return through the Space Station, and full capture of the MTV into LEO orbit.
Cryo/Aerobrake Mission Operations Outline

This is a top level outline of the major task sequences and their relative location for the Cryogenic fuel/Aerobrake vehicle for Mars missions. It is divided into four segments:

a) Near-Earth Operations (initial) - involving operations from hardware buildup to Trans-Mars Burn

b) Transit Operations - operations to be performed during the outbound transit flight from Trans-Mars Burn through the MTV/MEV separation for Mars atmosphere aerocapture

c) Mars Operations - covers the events from MEV and MTV aerocapture through the Trans-Earth Burn

d) Transit and Near Earth Operations - looks at the inbound transit return to Earth and capture operations at Earth

These segments will be further broken down into distinct top-level tasks in the Mars Operations Task Flow for the Cryo/Aerobrake.
Mars Major Mission Operations
Cryo-Aerobrake

Near-Earth Operations:
- Ground testing and support
- Launch support and Launch
- On orbit assembly and checkout
- Departure positioning and TMI burn

Transit Operations:
- TMI stage drop
- Transit flight configuration
- Optional maneuvers- Venus swingby, Deep Space Burn, coast correction and reconfigure
- Periodic Maintenance and inspection
- Autonomous checks and separation maneuvers

Mars Operations:
- Aerocapture, rendezvous and dock
- Separation
  - MEV: Land and establish base, ascend
  - MTV: Autonomous orbit, communications relay, survey
- Rendezvous and dock
- Jettison excess mass
- TEI burn

Transit and Near Earth:
- Transit configuration
- Optional Maneuvers- coast correction, Venus Swingby, reconfigure
- Earth Return- ECCV direct entry or ECCV orbit capture or MTV capture
Mars Mission Operations Task Flow
Cryo/Aerobrake

This is a task by task top-level operations task flow for the Cryo/Aerobrake to complete a general Mars mission. Those tasks that are optional (may or may not be used on any given mission) are shown with darkened background. Those operations flows that present alternatives to the baselined flow or are alternative actions are shown with dashed flow lines.

The assumptions under which these flows were developed are these: Communications with Earth is periodic, for the most part the decision making capability is with the astronauts beyond the TMI burn. This is due to the long time lag in communications (up to an hour) with Earth. Critical maneuvers such as flybys, Deep Space Burns, craft separations and docking, landing and ascent, and TEI burn will be reported to Earth prior to and after the event. Autonomous system self check capability will be present on all vehicle systems used for assembly and maintenance. Robotic assembly of the vehicle- MTV and MEV done at SSF/ on orbit, TMI assembly and integration, propellant top-off and final inspection and checkout done off-station. and the manned transfer vehicle is self-sufficient in repair capability.
Assumptions

- Communications with Earth is periodic, for the most part the decision making capability is with the astronauts beyond the TMI burn. This is due to the long time lag in communications (up to an hour) with the Earth. Critical maneuvers such as flybys, Deep Space Burns, craft separations and docking, landing and ascent, and TEI burn will be reported to Earth prior to and after the event.

- Autonomous system self check capability present on all vehicle parts used for assembly and maintenance

- Robotic assembly of the vehicle - MTV and MEV done at SSF/on orbit, TMI assembly and integration, propellant top-off and final inspection and checkout done off-station

- Manned transfer vehicle is self-sufficient in repair capability
Cryo/Aerobrake - Operating Modes and Options - Other

Presented here is a launch configuration for the externally mounted, fully assembled aerobrake. This option has been called “Ninja Turtle” launch configuration. While the initial investigation of launching the vehicle externally mounted indicated that “more work” had to be done to regard this as a viable option; some launch considerations such as shrouds and fairings were not considered in the original calculation which was based on a design sketch. The analyses also involved the launch of two aerobrakes instead of one and was, again, a preliminary analyses. We believe that this launch configuration deserves further analyses. It parallels the configuration of the Shuttle and would solve the problems of on-orbit construction of the aerobrakes and severely volume limited launches.
In manifesting the Mars vehicle we examined alternatives to launching the aerobrake in sections, which is a volume limited load. The result was this "turtle" or "piggyback" configuration that may be able to take additional payload up secured to the aerobrake and/or the inline section. This is still a preliminary design which must be analyzed for launch loads and aerodynamics.
Shuttle Derived Aerobrake Launch Option

- Aerobrake completely assembled on Earth.
- Eliminates in-orbit assembly/check-out.
- Eliminates packing problem.
- Uses shuttle derived launch vehicle.
- Can possibly pack MEV lander structure and descent tanks in aerobrake at launch.
- Recommended launch vehicle for Lunar/Mars initiative.
- 90t. launch capacity.

Diagram:
- Modified E.T.
- Standard SRB's
- (2) Aerobrakes
- (3) SSME's
IV. System Description of the Vehicle
IV. System Description

A. Part Descriptions

The first set of charts tabulates subsystem characteristics for the seven flight vehicle elements of the cryogenic/aerobrake vehicle. The following chart presents trade item decisions and rationale for subsystem choices.

The Mars excursion vehicle (MEV) is packaged into an asymmetrical aerobrake for Mars capture and landing. The shape of the brake and the configuration of the MEV are driven by the (assumed) 22 degree wake deflection inward of the velocity vector streamline. The other packaging consideration is placing the center of mass in line with the aerodynamic force vector. An alternative configuration includes a Mars surface reconnaissance (MSR) vehicle which is mounted on the aerobrake for Mars capture. The MSR vehicle lands at a different site than the manned lander, then returns a sample to the manned lander for return to Earth.

The Mars transfer vehicle and Mars excursion vehicle are docked together during the planetary transfers to gain the use of the combined volumes for crew habitation. A short duration crew module is used for return to orbit. It carries a crew of four. The MEV includes the crew module, descent and ascent stages, and a surface habitat. The MEV has a landing leg span of almost 20m and a height overall of 14m. Several views are provided from a computer solid model of the MEV.

An Earth crew capture vehicle is used for crew return to the Earth's surface. The configuration shown is for a crew size of five, although subsequent analysis indicates a crew of six is required to provide adequate crew skill redundancy.

Habitation Module Weight Trade Study. This study considered different module shapes for varying crew size, to determine least weight solutions. Primary and some secondary structure were considered in this study as weight discriminators. It was assumed that certain penetration-related secondary structure (airlocks, hatches, windows) and interiors (non-pressure bearing floors and walls, and equipment mounting) would not be significantly different in weight across the options. The results indicate larger diameter (i.e. more spherical) modules are lighter than the SSF design for large crew sizes.

The selected module concept has a 7.6m diameter and a 2:1 aspect ratio with elliptical end domes.

Cryogenic Boiloff Code Tank Estimation. Tank characteristics as a function of operating pressure and multilayer insulation thickness were estimated. The estimates generated by the computer model agree well with the actual mass of the Space Shuttle External Tank when the External Tank capacities are used.

Relative Development Effort Comparison. Estimates of the development effort for each propulsion element in a total Lunar/Mars program were made for various combinations of propulsion. The nuclear thermal rocket yielded the lowest effort estimate on a relative scale. This is only a gross comparison, not considering the differing cost of propulsion developments.
Subsystem Summary

The following three charts contain a concise listing of the primary characteristics of seven major subsystems for each of the seven major reference vehicle stages.
# Subsystem Summary

<table>
<thead>
<tr>
<th>Subsystems</th>
<th>MTV Crew Module</th>
<th>MTV TMI Stage</th>
<th>MTV TEI Stage</th>
<th>MEV Descent Stage</th>
<th>MEV Ascent Stage</th>
<th>MEV Crew Module</th>
<th>ECCV</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Structures</strong></td>
<td>7.6m Dia. x 9m L Al, axial tension tie internal bulkhead</td>
<td>Ribbed skin, Thrust struct., LOX/LH2 tanks, plumbing, etc. Prop. fract. = 0.9</td>
<td>Truss tube, Thrust struct., LOX/LH2 tanks, plumbing, etc. Frame struct. 5% of TEI prop &amp; inert mass</td>
<td>Thrust struct, tanks, etc. Frame struct 10% of inert mass. Landing legs - 3% landed mass</td>
<td>Thrust struct, tanks with vac shell, etc.</td>
<td>4.4m D 0.5 ellipsoidal Al shell</td>
<td>3.9m x 2.7m Apollo type capsule</td>
</tr>
<tr>
<td><strong>Thermal Control</strong></td>
<td>Water/Glycol w/body mounted radiator integral w/meteoroid shield</td>
<td>Passive</td>
<td>Passive</td>
<td>Passive</td>
<td>Passive</td>
<td>Water/Glycol w/ ext. panel radiator &amp; water flash evaporator</td>
<td>Water/Glycol</td>
</tr>
<tr>
<td><strong>Aerobrake structures</strong></td>
<td>Rigid, Deep shell assembled in segments. Spar ribbed: Honeycomb face sheets. 13% cap mass.</td>
<td>N/A</td>
<td>N/A</td>
<td>Same dimensions as MTV aerobrake 13% of capture mass</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
</tr>
<tr>
<td><strong>Thermal</strong></td>
<td>Reradiative TPS</td>
<td>N/A</td>
<td>N/A</td>
<td>Reradiative</td>
<td>N/A</td>
<td>N/A</td>
<td>Ablative Shield</td>
</tr>
</tbody>
</table>
# Subsystem Summary (continued)

<table>
<thead>
<tr>
<th>Subsystems</th>
<th>MTV Crew Module</th>
<th>MTV TMI Stage</th>
<th>MTV TEI Stage</th>
<th>MEV Descent Stage</th>
<th>MEV Ascent Stage</th>
<th>MEV Crew Module</th>
<th>ECCV</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Avionics</strong></td>
<td>SSF derived command, ctrl. &amp; data handling equip., GN&amp;C platforms, new comm. systems, health monit., aerobrake attitude control</td>
<td>Main propulsion controls / instrumentation</td>
<td>Main propulsion C&amp;I solar array positioning control, RMS pos. ctrl. unit, Aeroshell integrity monitor</td>
<td>Main propulsion C&amp;I, Cryo prop monitor sys.,</td>
<td>Apollo/LEM type complete flight ctrl system Onboard health monitoring equip</td>
<td>Apollo Command Module type</td>
<td></td>
</tr>
<tr>
<td><strong>Power</strong></td>
<td>15 kW, solar arrays w/ battery storage</td>
<td>Distribution sys., post separation battery power</td>
<td>Distribution sys., Back-up fuel cells system</td>
<td>Distribution sys., Back-up fuel cells system</td>
<td>2.3 kW fuel cell for Des/Asc, solar arrays for surface</td>
<td>Battery storage</td>
<td></td>
</tr>
<tr>
<td><strong>Propulsion Engines</strong></td>
<td>N/A</td>
<td>5-200k lb adv. engines (ASE),w/eng. out cap., stage T/W=0.4, Large area ratio, Isp=475</td>
<td>3-34k lb w/engine out cap. Stage T/W=0.2, Large area ratio, Isp=475</td>
<td>4-34k lb w/engine out cap. Ext/ret nozzles. Isp=460</td>
<td>2-34k lb. w/engine out cap. Ext/ret nozzles. Isp=460</td>
<td>N/A</td>
<td>N/A</td>
</tr>
</tbody>
</table>
### Subsystem Summary (continued)

<table>
<thead>
<tr>
<th>Subsystems</th>
<th>MTV Crew Module</th>
<th>MTV TMI Stage</th>
<th>MTV TEI Stage</th>
<th>MEV Descent Stage</th>
<th>MEV Ascent Stage</th>
<th>MEV Crew Module</th>
<th>ECCV</th>
</tr>
</thead>
<tbody>
<tr>
<td>ECLSS</td>
<td>SSF derived with all resupplies and change-out equip. onboard. Closed on H2O and O2</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td>Open loop Apollo type</td>
<td>Open loop Apollo type</td>
</tr>
<tr>
<td>Crew Accom.</td>
<td>40-50 cubic meters habitat volume/person. Dedicated radiation shelter. SSF level of crew comfort; shower etc.</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td>54 cubic meters total habitat volume, Spartan crew accom. 3-day nominal occupancy</td>
<td>8 cubic meters Apollo type crew accom. 3-day nominal occupancy</td>
</tr>
</tbody>
</table>
Mars Mission Trades & Issues

Several trade items or issues are listed with the applicable selections for the reference vehicle given as well as the rationale behind the selection. The two boxed trade items are examined in more detail in subsequent charts.
# Mars Mission Trades & Issues

<table>
<thead>
<tr>
<th>Trade item</th>
<th>Selection</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>MEV</strong></td>
<td><strong>Selection</strong></td>
<td><strong>Rationale</strong></td>
</tr>
<tr>
<td>Propulsion:</td>
<td>2 stg, aeroshell dropped during desc.</td>
<td>Alternate split stg, very attractive eng commonality w M dep, min number of eng's w eng out cap, yet to be traded</td>
</tr>
<tr>
<td>Staging</td>
<td>30-34k lbf,</td>
<td></td>
</tr>
<tr>
<td>Engine thrust</td>
<td>2 asc, 4 desc</td>
<td></td>
</tr>
<tr>
<td>Number of Engines</td>
<td>Current design, low Pc, conservative Isp=460</td>
<td></td>
</tr>
<tr>
<td>Current vs ASE</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Propellant</td>
<td>Cryo O2/H2</td>
<td>Min mass even at 600 day surf stay Maximum passive insulation</td>
</tr>
<tr>
<td>Thermal insulation</td>
<td>MLI, vapor cooled shields, vac jacket-asc</td>
<td></td>
</tr>
<tr>
<td>Configuration:</td>
<td>Stretched ellipsoid, bi-level</td>
<td>Min mass, ease of egress, SSF derived Thrust thru CG w eng out</td>
</tr>
<tr>
<td>Asc cab</td>
<td>O2 top, H2 mideck-asc, desc on frame</td>
<td></td>
</tr>
<tr>
<td>Tank placement</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Cab power</td>
<td>Fuel cell,</td>
<td>3 day occupancy</td>
</tr>
<tr>
<td>Surface power</td>
<td>Solar array, fuel cell,</td>
<td>30 day requirement</td>
</tr>
<tr>
<td>TEI</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Propulsion</td>
<td>3 x 30-34k lbf eng's for M dep,</td>
<td>Common w MEV eng's, Min Delta V Proven hardware Min IMEO</td>
</tr>
<tr>
<td>Parking orbit</td>
<td>1 sol by 250 km periapsis</td>
<td></td>
</tr>
<tr>
<td>ECLSS</td>
<td>SSF derived, near 100% closure</td>
<td></td>
</tr>
<tr>
<td>Reuse/recovery</td>
<td>Crew return via ECCV, no reuse</td>
<td></td>
</tr>
<tr>
<td>TMI</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Propulsion</td>
<td>5 x 150 k lbf Isp=475 engines</td>
<td>Performance, eng delivered w tank set yet to be traded yet to be traded</td>
</tr>
<tr>
<td>propellant loading</td>
<td>tanked delivery</td>
<td></td>
</tr>
<tr>
<td>Staging/recovery</td>
<td>Single stg, no recovery</td>
<td></td>
</tr>
</tbody>
</table>

---

D15-100262
Mars Excursion Vehicle (MEV)

The MEV packs into the aerobrake as shown. The landing gear fold up, the descent engine nozzles retract, and the surface stairs fold up for packaging. The aerobrake acts as a heat shield during descent; 60 seconds prior to aerobrake separation, the engines extend through doors in the aerobrake and fire. The landing gear deploys, jettisoning the aerobrake before terminal descent.
MEV Aerobraking Constraints

- Combined vehicle mass center (MEV plus Aerobrake) has been positioned directly on the resultant force vector for an L/D of 0.5
MEV captures with lower L/D
- MSRV deployed after MEV capture
- MSRV performs independent entry and landing
- MEV entry with higher L/D
- IMEO penalty 14.3 t

20° RWA (landing L/D 0.5)
10° RWA (capture L/D 0.37)
Mars Excursion Vehicle (MEV)
MTV Hab Trade Weight Groundrules

Primary Structure  (trade discriminators)

- Pressure vessel
- Structure rings and ribs
- Pressure bulkheads (if any)

Secondary Structure  (trade discriminators)

- Inter-module tunnels (if any)
- Inter-module integrating structure (if any)
- Pressure hatches separating redundant volumes
- Meteoroid, debris and thermal protection (surface-area-based)

Secondary Structure  (not included; non-discriminators to first order)

- Airlocks
- Hatches associated with airlocks / EVA
- Windows
- Floors
- Walls
- Subsystem mounting standoffs
Module Concept Selection

The proportions of this module type do not approximate that of SSF modules until crew sizes of about 12 are reached. Beyond that point, it is useful to think of clustering these 7.6m-diameter modules together in simple topologies to extend the habitable domain, for surface bases as well as for large-crew in-space transportation systems.

Finally, it is important to remember that the nature of the trade study has led us to generate a quite conservative habitat concept, which although it combines features demonstrated to be advantageous, still reflects a rather limiting set of assumptions. As a next step, concepts should be considered which combine this reference module type with the smaller diameter module types which we still see as widely applicable throughout all phases of the HEI. For advanced applications, clusters which mix module types and sizes promise good accommodation of functional requirements as well as interesting and stimulating psychological environments.
Module Concept Selection

Selection

Modified Mg2-I concept family selected for further reference use in the STCAEM study, for

- concept development activities
- trade & sensitivity analyses
- more detailed habitat system definition

Major Features

- 7.6 m diameter
- 2:1 aspect ratio, unpenetrated end domes
- Cross - section, bisecting bulkhead
- Diametral tension - tie, deep floor
- Extensive commonality across architecture: g-field optimized

6 Crew Configuration
Module Concept Selection (2)

Functionality

- Unitary vessel minimizes leakage, parts count
- Permits wide variety of internal outfitting designs
- Diametral floor maximizes nominal floor area, facilitates weight-reducing tension tie
- Compact domain, good for access-time safety
- Best overall multi-floor efficiency in g-condition for a range of crew sizes
- Less wall area than smaller diameter; outfitting can compensate

Integration

- Minimizes orbital assembly operations required
- 7.6 m launch shroud likely available for early HEI
- Large crews can be accommodated through simple clustering
- Compact habitat facilitates aerobrake integration

Perception

- Survey results show technical people perceive larger diameter concepts as more spacious
- Barrel vault proportionately invariant with crew (module) size, better than dome
- Module width has better plan aspect ratio than smaller diameters
- Low intrinsic number of unique spatial units; outfitting can compensate
- Lowest score for circulation option boredom over long duration

Cost

- Lightest weight (transportation cost critical for exploration vehicles)
- Welded-metal technology feasible here, well-understood
- Prime opportunity for M&P improvements, however
- End dome complication less than for 10 m size
- Commonality in growth architectures more appropriate for surface system applications
Cryo Boiloff Code Tank Properties Prediction for ET Sized Tanks

Overall Tank Fraction vs. Design Pressure for ET Sized Tank-sets

Tank Mass vs. Design Pressure for ET Sized Tank

NTR Hydrogen Tank Fraction vs. MLI Thickness For Varying Tank Operating Pressures

NTR Hydrogen Tank Mass vs. MLI Thickness For Varying Tank Operating Pressure

D615-10026-2

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Comparison of Predicted Tank Masses & Fractions With the External Tank

**External Tank Data**

- Total Dry Mass = 35,425 kg
- Propellant Load = 719,112 kg
- LH2 Load = 102,618 kg
- LOX Load = 616,493 kg
- LH2 Tank Mass = 14,402 kg
- LOX Tank Mass = 5695 kg
- Tank Max. Operating Press:
  - LH2 - 34 psi
  - LOX - 22 psi

Overall Tankage Fraction = 4.7%

LH2 Tank Fraction = 12.3%
LOX Tank Fraction = 0.91%

LH2/LOX Overall Tank Fraction = 2.72%

**Boiloff Code Predictions**

Assumptions:
- MLI Thickness = 2"
- Diameter = 4.2 m
- Tank Shape - Cylindrical tank with $\sqrt{2}$ ellipsoidal endcones
- Ullage = 5%
- Propellant Mass = ET Propellant Loads
  - Mass includes vapor cooled shields,
    supports, and para-to-ortho
  - H2 converter, where appl.

<table>
<thead>
<tr>
<th>Results:</th>
<th>Tank</th>
<th>Pressure</th>
<th>Tank Mass</th>
<th>Tank Fraction (overall)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>H2</td>
<td>35 psi</td>
<td>13306 kg</td>
<td>11.5 % (2.6 %)</td>
</tr>
<tr>
<td></td>
<td>O2</td>
<td>40 psi</td>
<td>14521 kg</td>
<td>12.4 % (2.8 %)</td>
</tr>
<tr>
<td></td>
<td>H2</td>
<td>50 psi</td>
<td>16951 kg</td>
<td>14.2 % (2.99 %)</td>
</tr>
<tr>
<td></td>
<td>O2</td>
<td>30 psi</td>
<td>5239 kg</td>
<td>0.84 % (2.99 %)</td>
</tr>
<tr>
<td>* NTR value-25 psi x 2 for overpress.</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>* 2 x overpress. allowance</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Major Propulsion Element List for 2000-2030 HEI program

**Primary Objective:** Furnish a top level list of all major propulsion elements necessary to a 3 decade HEI total program entailing Lunar, Mars opposition (short stay) and Mars conjunction (long stay) missions.

**Secondary objective:** Considering four candidate vehicle combinations (differentiated by propulsion system choice, each of which might satisfy all the space transfer objectives of a comprehensive HEI program) roughly evaluate or 'score' the total development effort required to bring each propulsion system element/technology up to flight readiness. Having done so, sum all the element scores for each of the candidate vehicle combinations in order to ascertain which combination meets HEI program objectives with least overall propulsion systems development effort. The 4 candidates are listed below:

1. Chemical Lunar with chemical Mars opposition (zero-g) & conjunction (art-g, tether system)
2. NTR Lunar, NTR Mars opposition (zero-g) & Mars conj (art-g, vehicle rotation about its Cg, no tether)
3. Chemical Lunar, SEP Mars opposition (zero-g) & SEP conj (art-g, tether system)
4. Chemical Lunar, SEP Mars opposition (zero-g) & SEP conj (art-g, tether system)

**Scores: Primary list:** The all NTR set had lowest total propulsion element count of 5, that is, 5 distinct elements were identified. The chemical/SEP combination followed with 8 elements, all chemical with 8, and chemical/NEP with 11. Differences in opinion as to what constitutes 'major' and 'distinct' propulsion elements might lead to slight variations in the totals, all depending on who does the counting.

**Scores: Secondary list:** The all NTR set scored the lowest in total propulsion elements development effort with a score of 13. The chemical/SEP combination and the all chemical set were about even with scores of 18 & 19 followed by chem/NEP at 27. These scores are relative, and only show how the 4 vehicle sets compare to one another; They are also subjective, and the differences in overall scores may be more pronounced, less pronounced or even change in rank depending on who is doing the evaluating. These rankings are not presented herein as the results of a precise technical trade study, but rather the results of a rough comparison 'methodology' with its major emphasis on a top down viewpoint in contrast to an analysis which as its emphasis on optimizing and/or selecting propulsion systems solely for individual missions.
Major Propulsion Element List for Specific Vehicle Sets
to Satisfy Lunar & Mars Objectives of 2000-2030 HEI Program

<table>
<thead>
<tr>
<th>Moon</th>
<th>zero-g opposition</th>
<th>Mars</th>
<th>artificial-g conjunction</th>
<th>Propulsion Element</th>
<th>Development Effort Factor</th>
</tr>
</thead>
<tbody>
<tr>
<td><img src="image1" alt="Moon Diagram" /></td>
<td><img src="image2" alt="zero-g opposition Diagram" /></td>
<td><img src="image3" alt="Mars Diagram" /></td>
<td><img src="image4" alt="artificial-g conjunction Diagram" /></td>
<td><strong>Lunar</strong></td>
<td><strong>Lunar</strong></td>
</tr>
<tr>
<td>1. LTV propul stag</td>
<td>2. LTV aerocapt brake</td>
<td>3. LEV propul stag</td>
<td><strong>2</strong></td>
<td><strong>Lunar Chemical/Mars Chemical sys</strong></td>
<td></td>
</tr>
<tr>
<td><strong>Mars zero-g vehicle</strong></td>
<td>4. MEV propul stag</td>
<td>5. MEV/MTV aerocapture brake</td>
<td>6. MTV propul stage</td>
<td>7. TMI propul stage</td>
<td><strong>2</strong></td>
</tr>
<tr>
<td>8. Art-g tether system</td>
<td>8. distinct propulsion elements</td>
<td>with development factor scores:</td>
<td><strong>19</strong></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

**Lunar NTR/Mars NTR sys**

<table>
<thead>
<tr>
<th>Lunar</th>
<th><strong>Common Lunar &amp; Mars zero-g</strong></th>
<th><strong>Mars artificial-g</strong></th>
</tr>
</thead>
<tbody>
<tr>
<td>1. LEV propul stage</td>
<td>2. Common LTV/MTV NTR propul stage</td>
<td>no necessary additions</td>
</tr>
<tr>
<td>3. Radiation handling/monitoring/shield</td>
<td>4. MEV propulsion stage</td>
<td>5. MEV descent heat shield</td>
</tr>
<tr>
<td><strong>2</strong></td>
<td>6</td>
<td>1</td>
</tr>
</tbody>
</table>

5 distinct propulsion elements
with development factors scores: **13**

**Legend:** (1) least development effort; (6) most development effort

Expected total resources that must be expended for such a propulsion element to achieve flight readiness.
Major Propulsion Element List for Specific Vehicle Sets to Satisfy Lunar & Mars Objectives of 2000-2030 HEI Program

<table>
<thead>
<tr>
<th>ADVANCED CIVIL SPACE SYSTEMS</th>
<th>BOEING</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Lunar</strong> zero-g opposition</td>
<td></td>
</tr>
<tr>
<td><strong>Mars</strong> artificial-g conjunction</td>
<td></td>
</tr>
<tr>
<td>Propulsion Element</td>
<td>Development</td>
</tr>
<tr>
<td><strong>Lunar</strong></td>
<td></td>
</tr>
<tr>
<td>1 LTV propulsion stg</td>
<td>Lunar</td>
</tr>
<tr>
<td>2 LTV aerocapture brake</td>
<td></td>
</tr>
<tr>
<td>3 LEV propulsion stg</td>
<td>NEP</td>
</tr>
<tr>
<td><strong>Mars</strong> zero-g vehicle</td>
<td></td>
</tr>
<tr>
<td>4 MEV propulsion stg</td>
<td>2</td>
</tr>
<tr>
<td>5 MEV descent heat shield</td>
<td>1</td>
</tr>
<tr>
<td>6 NEP reactor</td>
<td>6</td>
</tr>
<tr>
<td>7 Radiation handling/monitoring/shield</td>
<td>2</td>
</tr>
<tr>
<td>8 Dynamic conversion equip</td>
<td>2</td>
</tr>
<tr>
<td>9 Radiators</td>
<td>2</td>
</tr>
<tr>
<td>10 Electric thrusters</td>
<td>3</td>
</tr>
<tr>
<td>- Separate crew carrier to NEP 'spiral up' altitude</td>
<td></td>
</tr>
<tr>
<td><strong>Mars</strong> Artificial-g vehicle</td>
<td></td>
</tr>
<tr>
<td>11 Artificial-g tether system</td>
<td>2</td>
</tr>
<tr>
<td><strong>Lunar</strong></td>
<td></td>
</tr>
<tr>
<td>1 LTV propulsion stg</td>
<td>Lunar</td>
</tr>
<tr>
<td>2 LTV aerocapture brake</td>
<td></td>
</tr>
<tr>
<td>3 LEV propulsion stg</td>
<td>2</td>
</tr>
<tr>
<td><strong>Mars</strong> zero-g vehicle</td>
<td></td>
</tr>
<tr>
<td>4 MEV propulsion stg</td>
<td>2</td>
</tr>
<tr>
<td>5 MEV descent heat shield</td>
<td>1</td>
</tr>
<tr>
<td>6 SEP Solar array</td>
<td>3</td>
</tr>
<tr>
<td>7 Electric thrusters</td>
<td>3</td>
</tr>
<tr>
<td>- Separate crew carrier to SEP 'spiral up' altitude</td>
<td></td>
</tr>
<tr>
<td><strong>Mars</strong> Artificial-g vehicle</td>
<td>0 (use LTV)</td>
</tr>
<tr>
<td>8 Artificial-g tether system</td>
<td>2</td>
</tr>
<tr>
<td>8 elements w sum of devel factors scoring:</td>
<td>18</td>
</tr>
</tbody>
</table>

Legend: (1) least development effort; (6) most development effort
Expected total resources that must be expended for such a propulsion element to achieve flight readiness
Weights Statement
B. Weight Statements

Summary and detailed weight estimates are provided for the Chemical/aerobrake vehicle for the 2015 opposition mission opportunity. Assumptions made in the weight estimates include:

- Crew size of 4
- Use of Earth capture crew return vehicle
- Mission duration of 565 days.
- Improved technology (post-1990) for component weights (see technology section). The reference mass for this mission case is 800 tons in low Earth orbit.
Reference Chem/aerobrake Vehicle for 2015 Opposition Mission
Crew of 4, ECCV return, 565 day total trip time  Revision 2 5/22/90

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>MTV Mars aerobrake</td>
<td>23758</td>
</tr>
<tr>
<td>MTV crew hab module 'dry'</td>
<td>28531</td>
</tr>
<tr>
<td>MTV consumables &amp; resupply</td>
<td>7096</td>
</tr>
<tr>
<td>MTV science</td>
<td>1000</td>
</tr>
<tr>
<td>MTV propulsion stage</td>
<td>18206</td>
</tr>
<tr>
<td>MTV propellant load</td>
<td>85141</td>
</tr>
<tr>
<td>MTV total</td>
<td>163732</td>
</tr>
<tr>
<td>MEVMars capture &amp; desc aerobrake</td>
<td>15138</td>
</tr>
<tr>
<td>MEV ascent stage</td>
<td>22754</td>
</tr>
<tr>
<td>MEV descent stage</td>
<td>21457</td>
</tr>
<tr>
<td>MEV surface cargo</td>
<td>25000</td>
</tr>
<tr>
<td>MEV total</td>
<td>84349</td>
</tr>
<tr>
<td>ECCV</td>
<td>7000</td>
</tr>
<tr>
<td>Cargo to Mars orbit only</td>
<td>0</td>
</tr>
<tr>
<td>MTV-TMI Interstage wt</td>
<td>500</td>
</tr>
<tr>
<td>TMI inert stage wt</td>
<td>54560</td>
</tr>
<tr>
<td>TMI propellant load</td>
<td>490950</td>
</tr>
<tr>
<td>TMI stage total</td>
<td>545510</td>
</tr>
</tbody>
</table>

**IMLeo**

801090

Mac chart: M Ref Chem/ab cover pg
synthesis model run# marschemmtv.dat,21
# Ascent Cab - Ref MEV for 2015 Chem/Aerobrake Vehicle

_Crew of 4, 3 day occupancy time_  Revision 2 5/22/90

### Cab ECLSS

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>Atmospheric Revitization Sys/Trace contaminant control assembly</td>
<td>123</td>
<td>CO2 adsorption unit, expendable LIOH cartridge, pre &amp; postsorbent beds, catalytic oxidizer for particulate &amp; contaminant control</td>
</tr>
<tr>
<td>Atmosphere Control System</td>
<td>62</td>
<td>Total &amp; partial press control; valves, lines &amp; resupply/makeup O2 &amp; N2 and tanks</td>
</tr>
<tr>
<td>Atmos. Composition &amp; Monitor Assem.</td>
<td>55</td>
<td>O2 &amp; N2 monitor for ACS, particulate &amp; contaminant monitor for ARS</td>
</tr>
<tr>
<td>Thermal Control Sys</td>
<td>40</td>
<td>Temp control: sensible liq. heat exchanger, ext radiator wt included in 'secondary structure' mass</td>
</tr>
<tr>
<td>Temp. &amp; Humidity Control</td>
<td>240</td>
<td>Condensing heat exchanger, fans, ducting</td>
</tr>
<tr>
<td>Water Recovery and Management</td>
<td>45</td>
<td>Stored Potable water only</td>
</tr>
<tr>
<td>Fire Detection &amp; Suppression Sys.</td>
<td>113</td>
<td>Automatic sys w manual extinguishers as backup</td>
</tr>
<tr>
<td>Waste Management Sys and Storage</td>
<td>-</td>
<td>Considered part of 'Man Systems'</td>
</tr>
<tr>
<td><strong>Asc cab ECLSS mass</strong></td>
<td><strong>678</strong></td>
<td>Apollo style open ECLSS system</td>
</tr>
</tbody>
</table>

### Cab Structure

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>Primary/Secondary Structure</td>
<td>519</td>
<td>Overpressurized (20 psia) on launch for structural integrity.</td>
</tr>
<tr>
<td>Berthing ring/mecanism (1)</td>
<td>139</td>
<td>Stiffening rings added at cylinder/encap interface for added strength.</td>
</tr>
<tr>
<td>Berthing interface plate (1)</td>
<td>90</td>
<td>Skylab derived triangular grid floor with beam supports on 6&quot; centers.</td>
</tr>
<tr>
<td>Windows</td>
<td>90</td>
<td>Support ring interface on pressure vessel to carry loads imposed by the floor and equipment during launch to aerocapture.</td>
</tr>
<tr>
<td>Couches</td>
<td>80</td>
<td></td>
</tr>
<tr>
<td>Hatches (2)</td>
<td>80</td>
<td></td>
</tr>
<tr>
<td><strong>Asc cab Structure mass</strong></td>
<td><strong>998</strong></td>
<td></td>
</tr>
</tbody>
</table>

*synthesis model run number: marsmr.dat*

Mac chart: MEVasc cab wt-ratio
Asc stage - Reference MEV for 2015 Chem/Aerobrake Vehicle
Crew of 4, 30 day stay, 2 advanced space engines; Isp=475 sec  Revision 2 5/22/90

Structure  998 SSF dia center cyl section w ellip ends. Stiffening rings added. See 'Structures pg'
ECLSS  678 Open sys: CO2 adsorption unit, stored H2O,O2,N2, no airl., no hyg w. see 'ECLSS pg'
Command/Control/Power  330 Power: fuel cells
Man systems  82 Waste management sys/waste storage/medical equip.
Spares & tools  192 Subsystem component level spares
Wt growth  376 15% growth for dry mass
Asc 'dry' mass  2656 Total cab dry mass
Consumables (food & water)  62 Minimum; food and water only; 3 occupancy
Crew/effects/EVA suits  260 Crew of 4, 100 kg EVA suit per crew member
Ascent cab gross mass  3478

Fuel / Oxidizer

[43/46] Single tank wt  312/140 2 SIC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
[68/69] Meteoroid shield  40/18 One 0.40 mm sheet of Al
[50/51] MLI  59/26 MLI; density = 32 (kg/m3); 100 layers at 20 layers/cm.
[71/121/113] VCS & Vacuum shell  47/21 1 VCS and 1 Vac shell: both 2 x 0.13mm Al outer sheet w 0.57kg/m2 honeycomb core
[2x1316] Propel line wt  50/50 50 kg per tank
[116/117] Tank wt growth  62/21 15% wt growth
[73/74] Sum single tank inert  617/307 Total tank & tank inert wt

[500] Main propulsion  564 3 x 30kIbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
[118] Asc frame & struc wt  478 3% of total asc stage propulsion wt
[1274/525] RCS inert  122 Estimate from RCS prop load
[54] Propul, frame wt growth  174 15% of total inert
Sum Asc propul & frame inert  1338

[60] Asc usable propellant  15500 Asc veh dv = 5319 (m/sec) to 250 km periapsis alt. by 1 sol orbit.
[56-58] Asc boloff  418 50 day sep from MTV before M arr + 30 day surf stay; calc: Boeing CRYSTORE program
[52] Asc RCS prop  172 N2O4/MMH prop, Isp=280 sec, Asc RCS dv =35 (m/sec)
Sum Total Asc propellant load  16090

[63] Asc veh total mass  22754 all masses in kg

Mac chart: M Ref MEV asc veh wt-rationale  STCAEM/bbd/22May90

synthesis model run: marslander.dat;108
## Desc stage - Reference MEV for 2015 Chem/Aerobrake Vehicle

*Crew of 4, 30 day stay, 4 advanced space engines; Isp=475 sec, 25 t surf cargo*

**Revision 2  5/22/90**

### Fuel / Oxidiser

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass (kg)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Single tank wt</td>
<td>242/126</td>
<td>2 SIC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm</td>
</tr>
<tr>
<td>Meteoroid Shield</td>
<td>31/16</td>
<td>One 0.40 mm sheet of Al</td>
</tr>
<tr>
<td>MIL</td>
<td>47/24</td>
<td>MIL: density = 32 (kg/m3); 100 layers at 20 layers/cm.</td>
</tr>
<tr>
<td>Vapor Cooled Shields</td>
<td>37/19</td>
<td>1 VCS at 2 x 0.13mm Al outer sheet w/ 0.57 kg/m2 honeycomb core</td>
</tr>
<tr>
<td>Vacuum shell</td>
<td>0/0</td>
<td>Not on desc tanks</td>
</tr>
<tr>
<td>Propellant line wt</td>
<td>50/50</td>
<td>50 kg per tank</td>
</tr>
<tr>
<td>Tank wt growth</td>
<td>41/23</td>
<td>15% wt growth</td>
</tr>
<tr>
<td>Sum single tank inert</td>
<td>448/258</td>
<td>Total single tank + tank inert wt</td>
</tr>
<tr>
<td><strong>Tot. Fuel &amp; Ox tanks</strong></td>
<td>896/516</td>
<td>2 LH2 &amp; 2 LO2 tanks</td>
</tr>
</tbody>
</table>

### Main propulsion

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass (kg)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main propulsion</td>
<td>1127</td>
<td>4 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles</td>
</tr>
<tr>
<td>Asc frame &amp; struct wt</td>
<td>587</td>
<td>4% of desc stage sig wt + 2% of surf crew mod mass</td>
</tr>
<tr>
<td>Landing legs</td>
<td>1487</td>
<td>3% of total landed mass</td>
</tr>
<tr>
<td>RCS Inert</td>
<td>331</td>
<td>Estimate from RCS prop load</td>
</tr>
<tr>
<td>Propul, frame wt growth</td>
<td>490</td>
<td>15% of total inert</td>
</tr>
<tr>
<td><strong>Desc propul &amp; frame Inert</strong></td>
<td>4002</td>
<td></td>
</tr>
</tbody>
</table>

### Prop loads

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass (kg)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Desc usable Prop</td>
<td>13477</td>
<td>Desc propulsive veh dv = 931 (m/sec) from 250 km periapsis alt. by 1 sol orbit.</td>
</tr>
<tr>
<td>Desc bollof</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>Desc RCS prop</td>
<td>2566</td>
<td></td>
</tr>
<tr>
<td><strong>Total Desc propellant load</strong></td>
<td>16043</td>
<td>N2O4/MMH prop, Isp=280 sec, desc RCS dv=100 (m/sec)</td>
</tr>
</tbody>
</table>

### MEV aerobrake:

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass (kg)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Primary spar wt</td>
<td>2484</td>
<td>200ksi spar strength</td>
</tr>
<tr>
<td>Secondary spar wt</td>
<td>2596</td>
<td>22.5 inch spar depth</td>
</tr>
<tr>
<td>Honeycomb wt</td>
<td>6758</td>
<td>Note: 200ksi may require additional material technology development efforts</td>
</tr>
<tr>
<td>TPS wt</td>
<td>3300</td>
<td></td>
</tr>
<tr>
<td><strong>Total:</strong></td>
<td>15138</td>
<td></td>
</tr>
</tbody>
</table>

### Surface crew hab module

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass (kg)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Asc veh total mass</td>
<td>25000</td>
<td>Level II Requirement: surf modulw, surf scien &amp; surf stay consumables</td>
</tr>
<tr>
<td></td>
<td>22754</td>
<td>from 'Asc stage' wt statement page</td>
</tr>
</tbody>
</table>

### MEV mass

<table>
<thead>
<tr>
<th>Mass (kg)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>84349</td>
<td>All masses in kg</td>
</tr>
</tbody>
</table>

*synthesis model run#: marlander darn. 1*

STCAEM/mbbd/3May90  Mac chart: M Ref MEV decs veh wt-ratio
# Crew hab mod - Reference MTV for 2015 Chem/aerobrake Vehicle

**Crew of 4, 565 day total trip time**  Revision 2  5/22/90

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>[360] Structure</td>
<td>8351</td>
<td>Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties</td>
</tr>
<tr>
<td>[363] ECLSS</td>
<td>4256</td>
<td>SSF derived with same degree of closure, sized for crew of 4 for 565 days</td>
</tr>
<tr>
<td>[364] Command/Control/Power</td>
<td></td>
<td></td>
</tr>
<tr>
<td>[364] • Internal</td>
<td>1159</td>
<td>ECWS, DMS, batteries, other avionics/computing/monitoring eq., conditioning equip</td>
</tr>
<tr>
<td>[368-316] Man systems</td>
<td>4121</td>
<td>Wts - all sys: SSF derived (as a funct. of crew size &amp; occupancy time) for Mars missions</td>
</tr>
<tr>
<td>[316] Crew &amp; effects</td>
<td>440</td>
<td>110 kg per person including personal belongings</td>
</tr>
<tr>
<td>[373] Spares/Tools</td>
<td>1496</td>
<td>Subsys component level spares. Life crit sys are 2 fault tolerant (approach of SSF)</td>
</tr>
<tr>
<td>[247] Radiation shelter</td>
<td>1802</td>
<td>Provides 10 g/cm² protection + 3-5 g/cm² provided by vehicle structure and equip</td>
</tr>
<tr>
<td>[377] Weight growth</td>
<td>2973</td>
<td>15% weight growth for dry mass excluding crew &amp; effects and radiation shelter</td>
</tr>
<tr>
<td>[378] Airlocks</td>
<td>1530</td>
<td>2 x 765 kg external airlocks (shuttle type airlocks modified for MTV mission)</td>
</tr>
<tr>
<td>[330] EVA suits</td>
<td>0</td>
<td>EVA suits weight counted in MBV ascent cab weight statement</td>
</tr>
<tr>
<td>[160] TTNC &amp; GN&amp;C platforms wt</td>
<td>863</td>
<td>3 platforms</td>
</tr>
<tr>
<td>Sum</td>
<td>28531</td>
<td>'dry' hab module represents structure and support systems equip &amp; hardware that are dependant on crew size and independant of mission duration</td>
</tr>
</tbody>
</table>

| [371] *On board equip resupply       | 1304      | Based on adjusted SSF resupply reqts for pot w, hyg w, ARS, TCS, THC & WMS |
| [398] *Consumables                   | 5792      | Crew of 4 for 565 days; food: 2.04 kg/man/day, food pkg: 0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man, food vol: 0.0055 m³/man/day, other: 0.0018. |
| [380] MTV crew mod 'wet' wt          | 7096      |                                                                           |

| [165] *Transfer science equipment     | 1000      | Inb and outb MTV science hardware and supplies                            |
| Remote Manipulator-arm Sys           | 0         | all large external self assembly hardware left in LEO                      |

| 380+179 MTV crew mod & support       | 36627     | This wt reflects the Boeing ref crew of 4 mod loaded for the 2015 opposition mission. The mod 'dry' wt represents a SSF type closed ECLS Sys (air >99%, water >95%) that serves the crew with 2 fault tolerance on all life critcal sys except structure. Its wt varies primarily with crew size. consumables wt varies with crew size and mission duration. |

* MTV hab mod consumables, resupply, and transit science dependant on mission duration, and free abort requirement. i.e. crew mod 'wet' wt will vary for different missions

Mac chart: M Ref MTV mod wt-rationale synthesis model run# marschemmtv.dat;21
Mars departure stg - Reference MTV for 2015 Chem/Aerob Veh

Crew of 4, 2 advanced space engines; Isp = 475 sec

Revision 2 5/22/90

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>[154] Fuel tank</td>
<td>5424</td>
<td>2 SIC/AI metal matrix tanks, 37 ksf working stress, tank MEOP=175kPa</td>
</tr>
<tr>
<td>[155] Oxygen tank</td>
<td>3100</td>
<td>2 SIC/AI metal matrix tanks, 37 ksf working stress, tank MEOP=175kPa</td>
</tr>
<tr>
<td>[158] MLI/meteor shield</td>
<td>1082</td>
<td>MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm. Meteor Shield: 2 (kg/m2)</td>
</tr>
<tr>
<td>[159] Frame structure</td>
<td>5132</td>
<td>5% of MTV propellant + 5% of MTV stg inert mass</td>
</tr>
<tr>
<td>[518] Main propulsion</td>
<td>794</td>
<td>2 x 30k lbf advanced space eng’s: Isp=475 s, high AR nozzle not extendible</td>
</tr>
<tr>
<td>[183] RCS inert</td>
<td>300</td>
<td>Scaled from RCS propellant</td>
</tr>
<tr>
<td>[160] Mass growth</td>
<td>2374</td>
<td>15% growth for inert stage</td>
</tr>
<tr>
<td>[161] Mars dep stg 'dry' wt</td>
<td>18286</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>TEI stage inert</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>[118] MTV RCS propellant</td>
<td>699</td>
<td>Storable: N2O4/MMH propellant, Isp=280 sec, MTV RCS dV=30 m/sec</td>
</tr>
<tr>
<td>[122] MTV inb midcourse burn prop</td>
<td>1256</td>
<td>delta V: 90 (m/sec); burn done with MTV Mars dep main propulsion</td>
</tr>
<tr>
<td>[128] Mars dep usable prop</td>
<td>71252</td>
<td>LH2/LO2, MR=6:1, Mars dep dV: 3400 m/sec usable=prop req after outb &amp; inorbit bolloff; 30 day bolloff period; calculated with Boeing's 'CRYSTORE' program</td>
</tr>
<tr>
<td>[545-546] In orbit Mars dep prop bolloff</td>
<td>426</td>
<td></td>
</tr>
<tr>
<td>[554] tot onboard prop at Mars arr</td>
<td>73986</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>TEI prop, boll-off &amp; totals</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>[556] Tot M dep propulsive stg wt</td>
<td>103347</td>
<td></td>
</tr>
</tbody>
</table>

( at time of E dep burn )

389

synthesis model run #: marschemmtv.dat;21
Mac chart M Ref MTV vch wt rational-
## TMI stg - Reference MTV for 2015 Chem/Aerobrake Veh

**ECCV Return, 4 x 200k lbf advanced space engines; Isp = 475 sec**

*Revision 2 5/22/90*

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>[556] Tot MTV Mars dep stg</td>
<td>103347</td>
<td>See mars dep stage wt statement</td>
</tr>
<tr>
<td>[380+179] MTV Crew hab mod sys</td>
<td>36627</td>
<td>See MTV crew hab module wt statement</td>
</tr>
<tr>
<td>[230] ECCV</td>
<td>7000</td>
<td>4 man apollo type entry vehicle; MTV expended</td>
</tr>
<tr>
<td>[106] MEV</td>
<td>84349</td>
<td>4 man, 30 day stay, 25 t surface cargo</td>
</tr>
<tr>
<td>[159] Outb 'to-Mars-orbit' cargo</td>
<td></td>
<td>0 communication sat's taken on precursor mission</td>
</tr>
<tr>
<td>[1292] Mars Site Recon Vehicle</td>
<td>0</td>
<td>Not taken for Ref 2015 mission</td>
</tr>
<tr>
<td>[163] MTV-TMI interstage wt</td>
<td>500</td>
<td>Structural member joining TMI to MTV</td>
</tr>
</tbody>
</table>

### MTV Pay Load

**MTV Mars capture aerobrake:**

- Primary spar weight: 4239, 200ksi spar strength
- Secondary spar wt: 3434, 22.5 inch spar depth
- Honeycomb wt: 12785, note: 200ksi may require additional material technology development efforts
- TPS wt: 3300

**Total:** 23758

### TMI Stage

**Tot TMI stg 'Payload wt':** 255581

TMI propulsive stg injects this wt into hyperbolic trajectory

**TMI stage inert** 54560, 0.9 propellant fraction

**TMI propellant load** 490950, TMI stage tanks topped off before ignition, no boff account for

**TMI stage total mass** 545510, 4 x 200k lbf advanced space engines, Isp=475 sec

**IMLEO**

*initial mass in low Earth orbit* 80190

---

synthesis model run #: marschemmtv.dat;
Mac chart M Ref TMI wt-rationale
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Artificial Gravity Option
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Cryogenic/Aerobrake Vehicle
Artificial Gravity Configuration

The cryo/ab artificial gravity configuration employs a tether to achieve the radius desired to spin the transfer habitat at 56 m and 4 rpm to produce 1g. The tethers used are conductive tethers to avoid having separate power lines running in conjunction with the tether, thus complicating the reeling cycles. The conductive tether used is "ribbon" shaped to avoid entanglement during the reeling cycles, to better facilitate "crawler" operations, and because it radiates conductive heat better due to increased surface area over a circular cross-section.

The configuration is a 3 tether planar beam configuration with the crawler, solar arrays and communications laser located at the CM. The vehicle separates post-TMI with the transfer hab and MEV contiguously connected and the MTV aerobrake and TEI propellant used as countermass. The Mars to Earth configuration uses the MTV aerobrake and the empty TEI propellant tanks as countermass which results in a longer countermass radius to keep the transfer habitat at 56 m. If the MTV aerobrake is jettisoned at Mars in a nonreusable scenario, the Mars to Earth countermass radius would increase substantially to over 2 km.

The crawler/mast/power configuration at the CM of the vehicle is deployed on trusses that package into the crawler assembly. The solar array and the communications laser are on despun joints for tracking, and the entire assembly packages below the transfer habitat in the MTV. The crawler is divided into 2 sections so that one section can always be at the CM to support the deployable truss and the tether. The crawler taps into the aluminum conductor to transfer power from the solar array to the crew systems. Each crawler section has 2 small solar arrays for independent power during movement along the tether.

The cryo/ab mass penalty, when compared to a reusable 0g version, is ~ 15%, because of the hardware and propellant required to support artificial gravity operations. The MTV aerobrake would have to increase in size from 30 m to 32 m to accommodate packaging of the tether reel, crawler, solar arrays, and communications laser below the transfer habitat. 2 despun joints are also required for the solar array and communications laser.
Artificial Gravity (gₐ) Assessment
Assumptions

A 1g gravity level was assumed for this study over partial g because the minimum gravity level required to offset physiological deterioration is not known. The rotation rate was set to be no more than 4 rpm, which is based on experimental data in the Pensacola Slow Rotation Room (1960's) on human adaptation. The crew compartments are contiguously pressurized during all mission phases, and the crew modules are to be oriented with the long axis parallel to the spin vector to offset the Coriolis effect along major circulation paths. Connections between habitation and the countermass are either tethers or a truss rather than a pressurized tunnel because, since all crew compartments are contiguous, there is no need for an IVA transfer.
# Artificial Gravity ($g_a$) Assessment

<table>
<thead>
<tr>
<th>Assumptions</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>1g gravity level</td>
<td>• Earth-normal conditioning for exploration in surface EMU</td>
</tr>
<tr>
<td>Rotation rate ≤ 4 rpm (56 m)</td>
<td>• Generally accepted range for vestibular disturbance tolerance</td>
</tr>
<tr>
<td>Contiguous crew compartments</td>
<td>• Maximize available volume</td>
</tr>
<tr>
<td></td>
<td>• In-flight simulation and training</td>
</tr>
<tr>
<td></td>
<td>• Contingency operations</td>
</tr>
<tr>
<td>Truss and tether connections</td>
<td>• Avoids mass penalty</td>
</tr>
<tr>
<td></td>
<td>• Not needed for contiguous volumes</td>
</tr>
<tr>
<td></td>
<td>• Facilitates conductors</td>
</tr>
<tr>
<td>Module orientation parallel to spin vector</td>
<td>• $g$ level consistency; minimizing vestibular disturbance</td>
</tr>
<tr>
<td></td>
<td>• Mass properties quasi-isotropic to first order</td>
</tr>
</tbody>
</table>

D615-100252

/STCAEMs 1/12June90
This chart shows the 3 main configurations of the vehicle in transit. The Earth to Mars phase requires a total tether length of 128m, while the Mars to Earth phase requires a total tether length of 161m with MTV aerobrake, and 2.15km without MTV aerobrake. The solar array and the communications laser are located at the CM on a "despun" joint to track the sun and Earth respectively. The crawler is also located at the CM nominally, but has the ability to travel to either end of the central tether to transfer crew and/or supplies. The initial TMI configuration is shown for comparison.
g_a Cryo/AB Configuration

**Earth to Mars Configuration**

MTV (propellant, tanks & aerobrake) "Crawler"

**TMI Configuration**

Solar Array
Transfer Habitat
MEV
Communications Laser
(3) Tethers

79m
56m

**Mars to Earth Configuration**

MTV (empty propellant tanks & aerobrake)

Tether Reel
Transfer Habitat

112m
56m
This chart shows the relative mass of the Cryo/AB and the NTR artificial gravity configurations as compared to all the reference 0g configurations. The Cryo/AB configuration trades very poorly in artificial gravity, whereas the NTR configuration has only minor mass impact.
** Mass Summary**

* ECCV crew return, MTV aerobrake not used as return countermass → Level II 2015 565d option
** Earth aerocapture, MTV aerobrake used as return countermass
+ 1/3g NTR option → Boeing nominal 2016 434d option
++ 1g option
Artificial Gravity (g_a) Assessment
Assumptions

- Gravity level
  - 1g chosen over partial g (less than 1g)

- Rotation rate
  - \( \leq 4 \text{ rpm} \) (4 rpm at 56 m nominally)

- Crew compartments
  - contiguously pressurized throughout all mission phases

- Connection
  - truss and tethers rather than a pressurized tunnel
  - multiple tethers are used that are "ribbon" shaped in cross section

- Module orientation
  - long axis parallel to spin vector
**g_a Mass Summary**

**ADVANCED CIVIL SPACE SYSTEMS**

**BOEING**

![Bar Chart]

- **Propulsive**
  - 0g ga ga
  - 0g
  - 0g ga ga
  - NTR
  - NEP
  - SEP
  - GCR 120d
  - GCR 180d

* ECCV crew return, MTV aerobrake not used as return countermass ➤ Level II 2015 565d option
** Earth aerocapture, MTV aerobrake used as return countermass
# Represents a conjunction class mission, thus the reduced mass
+ 1/3g NTR option ➤ Boeing nominal 2016 434d option
++ 1g option
## Cryo/AB Mass Statement

### Advanced Civil Space Systems

**Element** | **ECCV return** | **Earth Aerocapture**
--- | --- | ---
MTV Mars aerobrake | 23758 | 23758
MTV crew hab module 'dry' | 28531 | 28531
MTV consumables & resupply | 7096 | 7096
MTV science | 1000 | 1000
MTV propulsion stage | 21847 | 27473
MTV MPS propellant load | 103129 | 132177
*MTV Art-g added hardware wt* | 8057 | 8057
*MTV Art-g added RCS prop* | 9004 | 10627
**MTV total** | 202422 | 238719

MEV Mars capture & desc aerobrake | 15138 | 15138
MEV ascent stage | 22754 | 22754
MEV descent stage | 21457 | 21457
MEV surface cargo | 23000 | 23000
**MEV total** | 84349 | 84349

**ECCV**
- Cargo to Mars orbit only | 7000 | 0
- MTV-TMI Interstage wt | 0 | 0
- TMI inert stage wt | 500 | 500

**TMI propellant load** | 62820 | 69080
**TMI stage total** | 565270 | 621540
**IMLEO** | 628090 | 690620

* Artificial -g system weight penalties all masses in kg

Earth aerocapture configuration shown in diagram

---

Mac chart: M 1-g chem/ab cover pg
synthesis modelrun#marschemmtv.dat; 33 & 34

/STCAEM/adc/06June90
**TMI stg - MTV for Artificial-g (1-g) 2015 Chem/Aerobrake Veh**

4 RPM, 4 spinup/dowm maneuvers, ECCV Return, 4 x 200k lbf adv eng's: Isp = 475

6/5/90

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tot MTV Mars dep stg</td>
<td>142037</td>
<td>See mars dep stage wt statement</td>
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<tr>
<td>MTV Crew hab mod sys</td>
<td>36627</td>
<td>See MTV crew hab module wt statement</td>
</tr>
<tr>
<td>ECCV</td>
<td>7000</td>
<td>4 man apollo type entry vehicle; MTV expented</td>
</tr>
<tr>
<td>MEV</td>
<td>84349</td>
<td>4 man, 30 day stay, 25 t surface cargo</td>
</tr>
<tr>
<td>Outb 'to-Mars-orbit' cargo</td>
<td>0</td>
<td>Communication sat's taken on precusor mission</td>
</tr>
<tr>
<td>Mars Site Recon Vehicle</td>
<td>0</td>
<td>Not taken for Ref 2015 mission</td>
</tr>
<tr>
<td>MTV-TMI Interstage wt</td>
<td>500</td>
<td>Structural member joining TMI to MTV</td>
</tr>
</tbody>
</table>

**MTV Mars capture aerobrake:**
- Primary spar weight: 4239
- Secondary spar wt: 3434
- Honeycomb wt: 12785
- TPS wt: 3300
- **Total:** 23758

| TMI Payload | | |
|-------------| | |
| Tot TMI stg 'Payload wt' | 294271 | TMI stage injects this 'payload' wt into Mars hyperbolic trajectory |

**TMI stage**
- TMI stage inert | 62820 | 0.9 propellant fraction |
- TMI propellant load | 565270 | TMI stage tanks topped off before ignition, no boiloff accounted for |
- TMI stage total mass | 628090 | 4 x 200k lbf advanced space engines, Isp=475 sec |

**IMLEO**
- Initial mass in low Earth orbit | 922361 |

---

Synthesis model run #marschemmtv.dat;33
Mac chart M 1-g TMI wt-rationale

/STCAEM/bbd/31May90
**Mars dep stg - for Artificial-g (1-g) 2015 Chem/Aerob Veh**

**4 RPM tether system, 4 spinup/down maneuvers, Crew of 4, 2 adv eng's; Isp = 475**

6/5/90

<table>
<thead>
<tr>
<th>Element</th>
<th>mass (kg)</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td>[154] Fuel tank</td>
<td>6545</td>
<td>2 Sic/AI metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa</td>
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<tr>
<td>[155] Oxygen tank</td>
<td>3769</td>
<td>2 Sic/AI metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa</td>
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<tr>
<td>[158] MLI/meteor shield</td>
<td>1282</td>
<td>MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm. Meteor shield: 2 (kg/m2)</td>
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<tr>
<td>[159] Frame structure</td>
<td>6243</td>
<td>5% of MTV propellant + 5% of MTV stg inert mass</td>
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<tr>
<td>[518] Main propulsion</td>
<td>794</td>
<td>2 x 30k lbf advanced space eng's; Isp=475 s, high AR nozzle not extendible</td>
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<tr>
<td>[183] Maneuver RCS inert</td>
<td>300</td>
<td>Scaled from RCS propellant</td>
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<tr>
<td>[160] Mass growth</td>
<td>2914</td>
<td>15% growth for inert stage</td>
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<tr>
<td>[161] Mars dep stg 'dry' wt</td>
<td>21847</td>
<td></td>
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</tbody>
</table>

Added wt necessitated by Art-g

- tethers: 2178 at 180 m each
- tether reel: 3050 Attached to MTV hab
- tether crawler: 1500 Transverses tether length, centers solar arr at veh/tether sys Cg for despun operation
- lock joints & other equip: 829 Lock joint secures MTV mod to MTV propulsion stg during MPS thrusting
- added Spin RCS sys: 500 Spinup/down RCS thrusters, lines, tanks, etc above the nominal maneuver RCS
- Total Art-g only hardware: 8057 Total Art-g hardware penalty not required for a zero-g vehicle

Art-g spin RCS propellant

- [224/225] outbound 1st spinup/down: 3298 Each of 2 counter wts (hab mod+MBV & MTV 'wet' propellant stg+AB) spin to 4 RPM
- [224/223] outbound 2nd spinup/down: 3258 Despun for outbound midcourse correction MPS burn, then respun to 4 RPM
- [222/221] inb 1st spinup/down: 1224 MBV left behind, MTV prop stg 'dry' except for inb midcourse correction burn
- [220/221] inb 2nd spinup/down: 1224
- Sum: 9004 Total Art-g RCS propellant penalty

RCS maneuver propellant

- [118] 599 Oaseous O2/H2 propellant, Isp=400 sec, MTV maneuver RCS dV=30 m/sec
- [122] delta V: 90 (m/sec); burn done w MTV Mars dep main propulsion
- [128] Mars dep usable prop: 87732 LH2/LO2, MR=6:1, Mars dep dV: 3400 m/sec usable=prop req after outb & inorb bolloff; 30 day bolloff period; calculated with Boeing's 'CRYSTORE' program
- [545+546] Outb midcourse burn prop: 485 midcourse maneuver delta V: 120 (m/sec); burn done w MTV main propulsion
- [121] 7638 335 day outbound trip time.
- [498+499] Outb Mars dep prop bolloff: 3129
- Sum: 103129 total MTV propellant load

<table>
<thead>
<tr>
<th>TEI prop &amp; bolloff</th>
</tr>
</thead>
<tbody>
<tr>
<td>[556] tot M dep propulsive stg wt (at time of E dep burn)</td>
</tr>
</tbody>
</table>

synthesis model run #marschemmmtv.dat:33

Mac chart M 1-g MTV veh wt-rationale

407
Cryo/AB Vehicle Features

- Nominal spin rate = 4 rpm (56m to create 1g)
- Conductive tether
- Sun-tracking solar arrays
- "Crawler" contingency for crew transfer from end to end
- Nominally 4 spin-up/spin-down cycles (1 for conjunction class mission)
  - Outbound - MTV aerobrake and propulsion as countermass
  - Inbound - Empty MTV propulsion and aerobrake as countermass (MEV expended)
    - MTV aerobrake not required because of ECCV; however, it is useful as a countermass and will be retained for fully-reusable mission modes
g. Cryo/AB Penalty Assessment

- Added mass
  - (3) tethers
  - Tether reel
  - Tether crawler
  - Added solar array
  - Added communications laser
  - Lock joints for transfer hab
  - Added RCS and propellant
  - Added TMI/TEI propellant

- MTV aerobrake
  - 2 m larger than MEV aerobrake
    - needed due to packaging constraints
    - complicates fabrication due to different sizes
  - Needed for inbound countermass - not needed in 0g option

- Spin-up/spin-down cycles
  - Mid-course correction problems

- "De-spun" joint for power and communication
ga Cryo/AB Tether Deployment Scenario

This chart outlines the vehicle deployment scenario. Omitted from this chart, for the interest of simplicity, are mid-course corrections, which would follow the same deployment scenario.
Cryo/AB Tether Deployment Scenario

Reference
- Post TMI, RCS fires to separate MTV (propulsion and AB) and MEV (+ transfer habitat) and deploy tether
  - tether slips freely through crawler
- Crawler clamps to CM point on conductive tether to finish deployment
  - 128m tether length outbound
  - 161m tether length inbound
- RCS fires to accelerate end masses to 4 rpm
- Crawler is positioned at CM and deploys solar array and communications laser
- Post Mars arrival, RCS fires to stop rotation
- Solar array/communications laser retract and crawler moves to MEV
- Tether is reeled in, maintaining slight tension
- RCS fires to slow approach to manageable STET speed
- Post berthing, Mars operations commence
- Reverse scenario after TEI using MTV AB and propellant tanks as countermass to transfer habitat

Alternative
- Deploy tether to twice intended length, small ΔV for rotation, then reel tether to nominal length
- saves propellant, but increases tether mass
This chart shows a detail of the central "crawler/mast" in the deployed and collapsed configurations. The solar array and the communications laser deploy on a deployable truss to separate the tethers and form a planar "beam". In the collapsed configuration, the solar array and the communications laser fold-up, spin 90° and package below the transfer hab on the MTV.
g.a Cryo/AB "Crawler/Mast" Configuration

15 KW Solar Array
De-Spun "Alpha Joint"

Array Housing
Tether Crawler
Struct. / Power Tether

Deployed Configuration
Comm. Laser

Collapsed Configuration
Cryo/AB Packaging Configuration

This chart shows the packaging configuration for the solar array, communications laser, crawler, and tether reel. Due to aerobraking constraints, the MTV aerobrake has to be 2m larger than the MEV aerobrake, which will cause problems in fabrication commonality. The MTV has been designed so that the transfer module and the artificial gravity equipment can slip out to deploy the tether for spin-up.
g_a Cryo/AB Packaging Configuration
Tether Crawler Configuration

A detail of the tether crawler is shown on this chart. The crawler is divided into 2 sections so that one section can always be at the CM to support the deployable truss and the tether. The crawler taps into the aluminum conductor to transfer power from the power source to the habitation areas. Each crawler section has 2 small solar arrays for power during movement along the tether and 2 roller motors.
Tether Crawler Configuration

Deployable Truss

Crawler Solar Array

Crawler Section 2

Roller Motor

Crawler Traction Rollers

Power Tap/Feed

Sliding Tether Sleeve

Truss Storage

2 cm thick tether

Crawler Section 1
Conductive Tethers

- Transmitting electric power through long flexible cables is standard practice on Earth
  - technology issues well understood

- Maintaining electrical contact between the mobile "crawler" and the tether
  - similar to track lighting, sliprings, electric motors and generators,
    electric subway trains, and trolleys
  - a Solar Power Satellite concept (1980) incorporated a 5 GWe slipring
  - partially despun spacecraft use lower power sliprings regularly
  - SSF solar arrays will use sliprings to carry tens of kWe

- The Remote Manipulator System on SSF will be much like a tether crawler
  - exception is that it uses power rather than providing it
  - crawls along SSF truss, stopping periodically to plug into electrical outlets

- Technology demonstration in 1991 on Tethered Satellite System (TSS) Shuttle flight
  - conductive tether with plasma contactors for electrodynamic experiment
Conductive Tethers

Conductive tethers have been used in this study to simplify the power transmission method. Conductive tethers are not a simple technology as demonstrated by the examples given on the following chart. Conductive tethers also simplify the realing process because of the reduced number of cables.
Conductive Tether Properties

- Kevlar 29

- $1.2 \times 10^6$ N stress

- Safety factor = 1.5 (using 3 tethers)

- (3) 180m tethers
  - 161m nominal separation
  - 56m radius to transfer habitat

- "Ribbon" shaped cross section
  - to avoid entanglement during reeling/unreeling cycles
  - easier crawler operations
  - 20 cm$^2$ cross-sectional area
  - radiates conductor heat better due to increased surface area
V. Support Systems
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Support Systems for the Mars Cryo/Aerobrake Transfer Vehicle.

The support systems necessary for the Mars Cryo/Aerobrake Transfer Vehicle consist of the interrelated and interdependent tasks of ground, launch, and on-orbit processing. Ground processing tasks for the Mars Vehicle include interface identification and verification as well as integrated systems testing. As the interface diagrams show, each part of the Mars Vehicle is connected (mechanically, electrically, data-wise, and/or fluidwise) to almost every other part. Earth-to-orbit (ETO) launch processing is constrained by both ground and on-orbit considerations. These tasks include launch site preparation, integrating the payload (in this case, the pieces of the Mars Vehicle) with the Heavy Lift Launch Vehicle (HLLV), and manifesting. The scheduling of hardware to be launched is bounded on one side by the ground test and verification program and on the other side by the on-orbit assembly plan. The selection of Assembly Node and assembly means (robotic, EVA, mix, etc.) are part of this analysis. The systems, facilities, plans, and purposes for each of these three levels of support are included within and represent the magnitude of effort necessary before a Mars vehicle is actually ready to fly.

Ground Processing. The first level, ground-based operations, begins with the identification of system interfaces for the Cryo/Aerobrake Vehicle. Subsystem interfaces are to be performed by the manufacturer; however, once complete systems have been delivered to the launch site, it is planned to perform system to system integration in order to test and verify interfaces and system flight readiness. The recommended approach is to use flight hardware to the greatest extent possible during system test and verification. The ground processing flow to accomplish these interface tasks determine when each system(s) must be available and when each will be ready for launch. The generic ground process involves: (1) receiving and inspection of the system(s); (2) assembly of system to system; (3) verification of interfaces and testing for flight readiness; (4) disassembly of system from system; (5) storage of system for other subsequent interface tests; and (5) processing of system for launch.

Launch Processing. Launch processing and sequencing constitute the second level of support systems. Processing tasks include integrated assembly and checkout of Mars Vehicle systems with the ETO vehicle. One of the most significant impacts to the assembly and launch facilities as well as to the launch vehicle itself may be the option of launching the aerobrakes fully integrated (the "Ninja Turtle" concept). This concept holds promise for reducing on-orbit assembly problems but raises some processing and launch vehicle compatibility issues. Manifesting analyses are dependent not only upon the ground and on-orbit operations but also upon the selection of the ETO launch vehicle. Several manifesting scenarios have been studied for a variety of HLLVs. In the majority of cases, the limiting factor is found to be payload volume, not mass, capacity.

On-orbit Processing. On-orbit operations, the third level of support systems, pertains to the assembly (and, for reusable vehicles, the disassembly and refurbishment) of the Mars Cryo/Aerobrake Vehicle. The choice of Assembly Node includes factors such as location, robotic and man-tended capabilities, accessibility, micrometeoroid/debris protection, operating systems, and on-orbit storage. An on-orbit assembly analysis has been performed for the reference vehicle (with the added constraint that the aerobrakes must be assembled in space) based upon one possible assembly platform which may be suitable for the Cryo/Aerobrake vehicle. This platform was designed to solve two of the major problems with assembly of the vehicle in Low Earth Orbit (LEO): debris protection and aerobrake construction. The STS External Tanks serve as both protection and a base upon which assembly mechanisms, storage, and vehicle integration may be performed. This is not intended to be the final solution to these problems; rather, this study serves to show one
possible solution at one possible node. The resulting analysis indicates that the main delimiter in assembly time is the launch frequency of the ETO vehicle.
Shuttle Derived Launch Vehicle Approach
For Lunar/Mars Initiative

This is a MSFC chart showing the launch vehicles considered in Earth-to-Orbit launches. We have done manifesting scenarios for the reference line of vehicles. These scenarios are shown in the following two charts, indicating what is manifested, type of vehicle, the number of launches, and the estimated payload mass per launch for the first three missions.
Shuttle Derived Launch Vehicle Approach for Lunar/Mars Initiative

**Pre-Lunar**
- Shuttle-C
  - 2 x ASRM's
  - 5id ET
  - 3 x SSME's
  - 4.6m x 25m P/L Envelope
  - 71t P/L Cap.

**Lunar**
- Shuttle-C Blk I
  - 7.6m x 27m P/L Envelope
  - Mod. ET
  - 61t P/L Cap.

**Mars**
- 2 Stage III.L.V
  - 12.5m x 30m P/L Envelope
  - 4 x ASRM's
  - New LOX/LH Core Stage
  - 5 x SSME's
  - Recoverable P/A Module
  - 140t P/L Cap.

- 3rd Stage Common w/MTV
- 4 x SSME's Core Stg
- Recoverable P/A Module
- P/L Capability
- 3 Stage = 120t (7.6m dia P/L Envelope)
- 2 Stage = 84t (10m dia P/L Envelope)

**Reference**
- Shuttle-C

**Alternative**
- SDV (Inline)
## Mars Mission Vehicle Manifests by Year
### Shuttle C/Z

<table>
<thead>
<tr>
<th>Flight #</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
<th>5</th>
<th>6</th>
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<td>Mars Departure</td>
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<tr>
<td></td>
<td>Manifest</td>
<td>MTV aerobrake, assembly equipment</td>
<td>Crew systems, ECCV, structure, comsats, consumables</td>
<td>TEIS O2 propellant, tankage</td>
<td>TEIS H2 propellant, tankage &amp; engines</td>
<td>MEV aerobrake, assembly equipment</td>
<td>Ascent vehicle, descent vehicle, surface payload</td>
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<tr>
<td>Mass</td>
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<td>50t</td>
<td>91t</td>
<td>15t</td>
<td>12t</td>
<td>75t</td>
<td>91t each launch</td>
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</table>

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<tr>
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<th>2</th>
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<td>50t payload, descent stage for MEV1</td>
<td>MEV 2 aerobrake, assembly equipment</td>
<td>50t payload, descent stage for MEV 2</td>
<td>MEV1-2 &amp; TMIS inter-connect structure, Nav kit</td>
<td>TMIS engines and departure propellant</td>
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<tr>
<td>Mass</td>
<td>12t</td>
<td>70t</td>
<td>12t</td>
<td>70t</td>
<td>20t</td>
<td>91t each launch</td>
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</table>

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<tr>
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<tr>
<td></td>
<td>Manifest</td>
<td>MTV aerobrake, assembly equipment</td>
<td>Crew Systems, ECCV, structure, comsats, consumables</td>
<td>TEIS propellant tankage O2 &amp; H2</td>
<td>MEV aerobrake, assembly equipment</td>
<td>Ascent vehicle descent vehicle, surface payload</td>
<td>MTV - Earth departure propellant &amp; engines</td>
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<tr>
<td>Mass</td>
<td>41t</td>
<td>50t</td>
<td>69 t</td>
<td>12t</td>
<td>75 t</td>
<td>85 t each launch</td>
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</tr>
</tbody>
</table>
# Mars Mission Manifests - Cryo/aerobrake and NTR

**140 t HLLV**

HLLV: Shroud Size - 30 meters x 10 m dia., 140 t throw weight

<table>
<thead>
<tr>
<th>Flight #</th>
<th>Launch Vehicle</th>
<th>Cryo/Aerobrake Manifest</th>
<th>Mass (t)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>HLLV</td>
<td>MEV Aerobrake (Ninja Turtle), Descent stage, surface cargo, assembly equipment, consumables</td>
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<tr>
<td>1</td>
<td>HLLV</td>
<td>Ascent stage, interconnect structure (MEV-MTV), TEI departure stage</td>
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</tr>
<tr>
<td>2</td>
<td>HLLV</td>
<td>MTV Hab Module, ECCV, MTV Aerobrake (Ninja Turtle), interconnect structure (MTV-TMIS), assembly equipment, consumables</td>
<td>100</td>
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<tr>
<td>3</td>
<td>HLLV</td>
<td>TMIS tankage, Engine set, Structure, assembly, equipment, Top-off equipment</td>
<td>140</td>
</tr>
<tr>
<td>4</td>
<td>HLLV</td>
<td>TMIS tankage and propellant</td>
<td>119.7</td>
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<tr>
<td>5</td>
<td>HLLV</td>
<td>TMIS tankage and propellant</td>
<td>119.7</td>
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<tr>
<td>6</td>
<td>HLLV</td>
<td>TMIS tankage and propellant</td>
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<tr>
<td>7</td>
<td>HLLV</td>
<td>TMIS tankage and propellant</td>
<td>140</td>
</tr>
<tr>
<td>8</td>
<td>HLLV</td>
<td>TMIS tankage and propellant (wet tanks and top-off)</td>
<td></td>
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</table>

<table>
<thead>
<tr>
<th>Flight #</th>
<th>Launch Vehicle</th>
<th>NTR Manifest</th>
<th>Mass (t)</th>
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<tbody>
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<tr>
<td>2</td>
<td>HLLV</td>
<td>TMI Tank</td>
<td>140</td>
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<tr>
<td>3</td>
<td>HLLV</td>
<td>TMI Tank</td>
<td>140</td>
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<tr>
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<td>HLLV</td>
<td>Engine, Shield structure, Mars Arrival/Departure Tank</td>
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<td>Reactor Engine and Shield, Mars Arrival/Departure Tank</td>
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<td>6</td>
<td>HLLV</td>
<td>Propellant for top-off</td>
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<td>HLLV</td>
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</table>
**Mars Mission Manifests - SEP and NEP**

**140 t HLLV**

HLLV : Shroud Size - 30 meters x 10 m dia., 140 t throw weight

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<tr>
<th>Flight #</th>
<th>1</th>
<th>2</th>
<th>3</th>
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<tr>
<td>Launch Vehicle</td>
<td>HLLV</td>
<td>HLLV</td>
<td>HLLV</td>
<td>HLLV</td>
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</tr>
<tr>
<td>Manifest</td>
<td>MTV Hab, Array Deployment Mech., Communications, ACS, Avionics, PPU's, Power Distribution &amp; Control, Main Truss, Array Structure, Experiment Platforms, Assembly Equipment</td>
<td>Transfer Arrays (18 of 18), MEV Aerobrake (Ninja Turtle), Assembly Equipment</td>
<td>Descent Module, Ascent Module, Surface Payload, Main Truss</td>
<td>Propellant Tanks, Propellant, Radiators, Array Blankets (14 of 18)</td>
<td>Propellant tanks, Propellant, Thruster pods, Radiators, Array Blankets (4 of 18)</td>
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<tr>
<td>Mass</td>
<td>82.2 t</td>
<td>45.5 t</td>
<td>88.8 t</td>
<td>104.3 t</td>
<td>96.5 t</td>
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</table>

**SEP**

<table>
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<th>Flight #</th>
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<td>Launch Vehicle</td>
<td>HLLV</td>
<td>HLLV</td>
<td>HLLV</td>
<td>HLLV</td>
<td>HLLV</td>
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<tr>
<td>Manifest</td>
<td>MTV Hab, MEV aerobrake (Ninja Turtle), ACS, Power conditioners, Communications, Assembly Equipment</td>
<td>Descent Module, Ascent Module, Surface Payload, Auxiliary Radiator, Power distribution and control</td>
<td>2 Propellant tanks, 3 thruster pods</td>
<td>1 Main Radiator, 1 thruster pod, 3 propellant tanks, 1 Auxiliary Radiator</td>
<td>2 nd Main Radiator, Reactors, Shields, Turbo pumps, Misc.</td>
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<tr>
<td>Mass</td>
<td>85.1 t</td>
<td>118.1 t</td>
<td>115.1 t</td>
<td>132.6 t</td>
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### Mars Mission Manifests - Cryo/aerobrake and NTR
#### 84 - 120 t HLLV

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<tr>
<td>Launch Vehicle</td>
<td>HLLV 1</td>
<td>HLLV 1</td>
<td>HLLV 1</td>
<td>HLLV 1</td>
<td>HLLV 1</td>
<td>HLLV 2</td>
<td>HLLV 2</td>
<td>HLLV 1</td>
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<tr>
<td>Cryo/Aerobrake Manifest</td>
<td>MTV Hab Module, Surface Payload, Assembly Equipment</td>
<td>MEV Aerobrake (Ninja Turtle) Habitat refurbish/consumables, Assembly Equipment</td>
<td>Descent Module, Ascent Module, structure</td>
<td>MTV Aerobrake (Ninja Turtle), Habitat refurbish/consumables, Assembly Equipment</td>
<td>TEIS Propellant, consumables</td>
<td>TMI Propellant &amp; Engines</td>
<td>TMI Propellant and tanks</td>
<td>Top-off Propellant and equipment, Habitat refurbish/consumables</td>
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<td>Mass</td>
<td>78.8 t</td>
<td>40.1 t</td>
<td>49.1 t</td>
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<td>84 t</td>
<td>117.7 t</td>
<td>114.5 t</td>
<td>84 t</td>
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<td>HLLV 1</td>
<td>HLLV 1</td>
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<td>HLLV 1</td>
<td>HLLV 1 or 2</td>
<td>HLLV 1 or 2</td>
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<tr>
<td>NTR Manifest</td>
<td>Descent Module, Ascent Module, Surface Payload, MEV aerobrake (Ninja Turtle), Assembly Equipment</td>
<td>Mars Departure/Earth Arrival Structure, Main Truss, MTV Hab</td>
<td>Engine/Shield Structure, Mars Arrival/Departure Tank (1 of 2)</td>
<td>Mars Arrival/Departure Tank (1 of 2)</td>
<td>Earth Departure Tank (1 of 2, 69.3 t off-loaded)</td>
<td>Earth Departure Tank (1 of 2, 69.3 t off-loaded)</td>
<td>Off-loaded Propellant from Flight 3 &amp; 6</td>
<td>Off-loaded Propellant from Flight 6 &amp; 7, Reactor Engine and Shield</td>
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<tr>
<td>Mass</td>
<td>112.2 t</td>
<td>46.6 t</td>
<td>84.0 t</td>
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<td>84 t</td>
<td>84 t</td>
<td>79.5 t</td>
<td>84 t</td>
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# Mars Mission Manifests - SEP and NEP

## 84 - 120 t HLLV

**HLLV 1:** Shroud Size - 30 meters x 10 m dia., 84 t throw weight  
**HLLV 2:** Shroud Size - 30 meters x 7.6 m dia., 120 t throw weight

<table>
<thead>
<tr>
<th>Flight #</th>
<th>Launch Vehicle</th>
<th>SEP Manifests</th>
<th>NEP Manifests</th>
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<tr>
<td>1</td>
<td>HLLV 2</td>
<td>Descent Module, Ascent Module, Surface Payload, MEV Aerobrake (Ninja Turtle), Assembly Equipment</td>
<td>Descent Module, Ascent Module, Surface Payload, MEV Aerobrake (Ninja Turtle), Assembly Equipment</td>
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<td></td>
<td></td>
<td>Array Deployment Mech., Communications, ACS PPUs, Avionics, Main Truss (3 of 4), Power Distribution &amp; Control, Array Structure, Experiment Platforms</td>
<td>Power Distribution and Control and Structure, ACS, Power Conditioning, Communications, Avionics</td>
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<td></td>
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<td>Propellant &amp; Tanks (1 of 4), Radiators (2 of 2), Main Truss (4th of 4), Transfer Arrays (8 of 18), Thruster Pods (2 of 2)</td>
<td>Propellant and Tanks (1 of 5), Thruster pods (3 of 4)</td>
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<td></td>
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<td>112.2 t</td>
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<td>HLLV 1</td>
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<td>3</td>
<td>HLLV 1</td>
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<td>Main Cycle Radiators (2 of 2), Thruster pod, Propellant and Tanks (2nd of 5)</td>
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<td>4</td>
<td>HLLV 1</td>
<td></td>
<td>Auxiliary Radiators (2 of 2), Turbopumps/Thermal loops, Propellant and Tanks (3rd of 5), MTV Hab, Reactors &amp; Shields, Propellant and Tanks (last 2)</td>
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<td>5</td>
<td>HLLV 1</td>
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<td>HLLV 1</td>
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<td>7</td>
<td>HLLV 2</td>
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Optional Manifesting of the four vehicle options was developed for a medium and large class HLLV. The analysis was completed by using theoretical volumetric and total mass calculations.
HLLV Optional Manifesting

- Manifesting data will be generated by volumetric and mass total calculations
- Aerobrake(s) will be assembled on-orbit
- Deployable truss type mechanisms are feasible
- Manifesting assumed on-orbit assembly at LEO
# On-Orbit Assembly Analysis  
(HLLV Missions to LEO)

<table>
<thead>
<tr>
<th>Propulsion Option</th>
<th>ETO Size</th>
<th>10 meter Dia. 30 meter long 84-120 mT class</th>
<th>12.5 meter Dia. 30 meter long 140-160 mT class</th>
<th>13.75 meter Dia. 38 meter long/22 meter nose cone 200-250 mT class</th>
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</thead>
<tbody>
<tr>
<td>Cryo/Aerobrake (opposition class)</td>
<td>11 missions 73 mT average</td>
<td>8 missions 101 mT average</td>
<td>5 missions 162 mT average</td>
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<tr>
<td>Solar Electric Power (opposition class)</td>
<td>TBD</td>
<td>5 missions 87.2 mT average</td>
<td>2 missions 218 mT average</td>
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<tr>
<td>Nuclear Electric Power (opposition class)</td>
<td>TBD</td>
<td>5 missions 108.9 mT average</td>
<td>3 missions 181.5 mT average</td>
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</tr>
<tr>
<td>Nuclear Thermal Rocket (opposition class)</td>
<td>TBD</td>
<td>6 missions 122 mT average</td>
<td>4 missions 183 mT average</td>
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</table>
Requirements For Earth Orbit Support Facility

This is a listing of the groundrules and assumptions used to begin analyzing the sequencing and operations for an orbit assembly facility.

note: this is a point design study
Requirements for Earth Orbit Support Facility

Groundrules
- Multiple (ETO) flights will be used in assembly
- Line of sight communications are to be used
- Extensive use of robotic and telepresent systems will be made
- Minimal EVA activities

Assumptions
- On orbit propellant fueling or launched wet propellant tanks may be used
- Two RMS systems will be used in assembly
- On orbit spares will be 15% of vehicle active component weight and 20% of inactive weight
- Robotic software and sensors will allow supervisory human control
- Proximity operations will be viewed directly or by video
- Assembly schedule will be two (2) years or less
Support Requirements and Concepts
Orbital and Space Based Requirements
Summary - February 23, 1990
On-Orbit Assembly

- **Purpose**
  Define orbital and space-based support equipment, crew and facilities requirement/interfaces. By transportation element, for each scenario (Task 5-2).

- **Man Mars Vehicle Baseline**
  - Mars Excursion Vehicle
    - Aerobrake
    - Descent System
    - Ascent System
    - Mars Surface Payload
    - Mars Science Payload

- **Mars Transfer Vehicle**
  - Aerobrake
  - Trans Earth Injection System
  - Habitat Module

- **Trans Mars Injection System**
  - Core Stack
  - Propellant Tank Set (3 Tanks Baseline)
Groundrules/Assumptions

- Off-SSF assembly of MMV

- HLLV available for MMV launch
  - 4 flights per year
  - On-orbit stationkeeping (< 1 week)

- SSF-based OMV capable of maneuvering complete MMV subassemblies (i.e. MTV crew habitat)

- Maximize automation and robotics for assembly tasks

- MMV LEO departure date---Feb 2016

- MMV has high level of BIT/BITE
On-Orbit Assembly Baseline

MMV assembly at ET-derived MMV Assembly Platform (MAP)
  Constructed prior to MMV FEL
  MAP is self-supporting with power, control, debris protection capability

Crew required for internal subsystem checkout, critical inspections, contingency, repair

All MMV components have standard STS grapple fixtures

RMS capability at MAP
  PRMS—2 MAP mounted 30m arms
  RAMS—4-30m arms (2 on each aerobrake)
  PAS—2 DOF anchors to hold large subassemblies
  ASF—Fixed anchors to store components prior to assembly

MMV aerobrake TPS installed on ground except around field joints
  TPS around field joints installed by PRMS

MAP has line-of-sight communications with SSF

Crew accommodations
  MTV habitat module provides early crew quarters
  Crew transferred from SSF in ACRV/OMV when needed
  SSF resource node contains workstation for MAP local control
  SSF PLM/OMV used for resupply of consumables/crew provisions/MMV spares
Definitions

MAP---MMV Assembly Platform
MMV---Mars Mission Vehicle
SSF---Space Station Freedom
HLLV---Heavy Lift Launch Vehicle
OMV---Orbital Maneuvering Vehicle
ET---External Tank
FEL---First Element Launch
RMS---Remote Manipulator System
PRMS---Platform Remote Manipulator System
RAMS---Remote Aerobrake Manipulator System
PAS---Platform Anchor System
ASF---Assembly Storage Fixtures
SSCC---Space Station Control Center
MMCC---Mars Mission Control Center
STS---Space Transportation System
TDRSS---Tracking and Data Relay Satellite System
Orbital Debris Environment

- MMV debris environment will be 20-25 times worse than current SSF requirement
  - Orbital debris environment for SSF dated 1985
  - New environment dated April 1989---4-5 times greater flux
  - Later MMV launch date (2016)---5 times greater flux
  - Shorter MMV stay time (10yr/2yr)---reduced probability of impact
  - Will by further modified by LDEF data

- SSF requirement
  - 0.9955 probability of no penetration for each module for 10 years

- SSF debris shielding planned
  - 0.05 in Aluminum shield
  - 4.3 in spacing
  - 0.125 in Aluminum pressure wall
Orbital Debris Environment (Cont)

Percentage of debris particles in a 2 deg interval

Normalized Closing Angle Density Function
500 km altitude, average 1990's environment
Change Request Cumulative Flux Distribution

500 km altitude

1995
- - - 2000
- - - - 2005

debris diameter (cm)
## Off-SSF Assembly Node
### Assembly Platform vs Integral MMV Assembly

<table>
<thead>
<tr>
<th>Assembly Platform</th>
<th>Advantages</th>
<th>Disadvantages</th>
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</thead>
<tbody>
<tr>
<td></td>
<td>Available prior to MMV FEL</td>
<td>Separate vehicle to control (ground/proximity operations)</td>
</tr>
<tr>
<td></td>
<td>Large debris protected area</td>
<td>Additional launches required</td>
</tr>
<tr>
<td></td>
<td>Platform for mounting power, control, communications subsystems</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Space for parallel assembly tasks/temporary storage of MMV components</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Platform for additional RMS</td>
<td></td>
</tr>
<tr>
<td>Integral MMV Assembly</td>
<td>All required subsystems already available in some form</td>
<td>Flight hardware used for debris protection</td>
</tr>
<tr>
<td></td>
<td>Allows thorough checkout of subsystems prior to launch</td>
<td>Modification of flight power, control subsystems to control vehicle during</td>
</tr>
<tr>
<td></td>
<td></td>
<td>assembly phase</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Requires storage space at SSF</td>
</tr>
</tbody>
</table>

Assembly platform provides large protected assembly workspace with minimum impact to MMV flight hardware
ET Debris Shield Concept for MMV Assembly

External Tank
47 x 8.4m dia
"Raft" of 6 ET's provides a 47 x 50.4m shield
"Raft" of 12 ET's provides a 94 x 50.4m shield

MTV/MBVAerobrakes
30 x 27.4 x 7m thick
ET Debris Shield Concept for MMV Assembly

- Concept---"Raft" expended NSTS External Tanks on-orbit to form debris shield
  - Requires two rows of 6 ET's each
  - Needs power, guidance, attitude control, reboost subsystems
  - Tilt shield into velocity vector to reduce drag
  - Provides work platform for assembly tasks
  - Assembled by NSTS

- Alternatives
  - Bring up debris shield with MMV---150K lbs for SSF-equivalent shield
  - Use MMV aerobrakes as shields---risk flight hardware
ET Debris Shield Concept for MMV Assembly (Cont)

- Advantages
  - One-third less weight penalty to orbit than separate shield---50K vs 150K lbs
  - Provides greater protection than SSF shield design
  - Debris shield can be completed years before MMV assembly start
  - Provides experience with on-orbit assembly of large structures

- Disadvantages
  - 3-4000 lbs penalty for each STS flight to orbit ET---48K lbs for 12 tanks
    - 1000 lbs for vent, tumble valve, range safety system modifications
    - 2-3000 lbs for OMS propellant
  - Additional 2000 lbs allowed for connecting structure/fill shielding for gaps
  - Debris shield is not continuous
  - High orbital drag (can be flown tilted into velocity vector to minimize drag)
  - May require on-orbit containment of SOFI
    - SOFI may become ablated, charred during ascent
    - UV degradation on-orbit
    - Outgassing
  - Requires development of power, guidance, attitude control, reboost systems
ET Debris Shield Assembly Sequence

- First Orbiter/ET flight
  - First Orbiter/ET taken to MMV assembly orbit
  - Orbiter separates from ET, turns PLB toward ET
  - EVA or RMS attachment of EPS, GNC, RCS, C&T packages

- Subsequent Orbiter/ET flights
  - Orbiter/ET rendezvous with debris shield
  - Attach connecting structure to existing ET hardpoints
  - Orbiter separates from ET, attach ET to debris shield
  - Upgrade/relocate subsystems as debris shield buildup continues
  - Complete initial configuration of 6 ET's/final of 12 ET's

- First MMV assembly flight
  - HLLV/OMV rendezvous with ET debris shield/assembly platform
  - Upgrade subsystems as required for aging, damage
  - Install PRMS
  - Stow MMV components on assembly platform
  - Initiate MMV assembly

- Subsequent MMV assembly flights
  - HLLV/OMV rendezvous with ET debris shield/assembly platform
  - Stow MMV components
  - Continue MMV assembly
  - Upgrade/relocate subsystems as MMV buildup continues
  - Resupply consumables
MMV Manifesting

- Vehicle: HLLV (2 or 3 Stage)
  - Abilities - 2 Stage
    - 10M x 30M Payload Envelope
    - 84 ton capacity
  - Abilities - 3 Stage
    - 7.6M x 30M Payload Envelope (less 3rd Stage)
    - 120 ton capacity
- HLLV Mission One (2 Stage)
  - MTV Habitat Module
  - Mars Surface Payload
  - Assembly Platform Support Equipment
- HLLV Mission Two (2 Stage)
  - MEV Aerobrake Sections
  - MTV Habitat Module Refurbishment/Consumables
- HLLV Mission Three (2 Stage)
  - MEV Aerobrake Sections
  - Assembly Platform Support Equipment
- HLLV Mission Four (2 Stage)
  - MEV Lander Structure
  - Lander Legs
  - Descent System
  - Ascent Payload
  - Airlock
  - Stairs

- HLLV Mission Five (2 Stage)
  - MTV Aerobrake Sections
  - MTV Habitat Module Consumables

- HLLV Mission Six (2 Stage)
  - MTV Aerobrake Sections
  - Assembly Platform Support Equipment

- HLLV Mission Seven (2 Stage)
  - MTV Trans Earth Injection System
  - MTV Habitat Consumables
  - Assembly Platform Support Equipment

- HLLV Mission Eight (3 Stage)
  - MTV Propellant with Engines

- HLLV Mission Nine thru Eleven (3 Stage)
  - MTV Propellant
On-Orbit Assembly

MMY TOP ASSEMBLY

- Smallest unit of time is 1 man-hour
- 16 man-hours = 1 man-day of Assembly Duration

BASELINE DURATIONS:
- HLV Launch = .5 man-day
- HLV achieves stable orbit = .25 man-day
- OMV deploys from/to Freedom = .5 man-day
- OMV berths to components = .25 man-day
- Unstow and power up Robotics = .06 man-day
- Robotic verification = .12 man-day
- HLV deploys components = .06 man-day
- OMV transfers components = .25 man-day
- Robotic tasks = .06 man-day
- EVA/Robotic Contingency = .5 man-day
- Component Inspection = .12 man-day
- Component Test = .25 man-day
- Subassemblies to stand-by mode = .5 man-day
- Mechanical Fastening of components = .18 man-day
# On-Orbit Assembly

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<td>INITIAL MTV AERI</td>
<td>BRACE ASSEMBLY MISSION</td>
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</tr>
</tbody>
</table>
## On-Orbit Assembly

<table>
<thead>
<tr>
<th>Name</th>
<th>Earliest Start</th>
<th>Earliest Finish</th>
<th>Subproject</th>
<th>Days</th>
</tr>
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<tbody>
<tr>
<td>HABITAT MODULE AND SURFACE PAYLOAD ASSEMBLY MISSION</td>
<td>2/28/13</td>
<td>6/14/13</td>
<td>HLLV MISSION ONE</td>
<td>106</td>
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<tr>
<td>INITIAL MEV AEROBRAKE ASSEMBLY MISSION</td>
<td>6/14/13</td>
<td>9/28/13</td>
<td>HLLV MISSION TWO</td>
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<tr>
<td>FINAL MEV AEROBRAKE ASSEMBLY MISSION</td>
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<td>1/12/14</td>
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<tr>
<td>MEV DESCENT, ASCENT, SURFACE AND SCIENCE PAYLOAD ASSEMBLY</td>
<td>1/13/14</td>
<td>4/29/14</td>
<td>HLLV MISSION FOUR</td>
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<tr>
<td>INITIAL MTV AEROBRAKE ASSEMBLY MISSION</td>
<td>4/29/14</td>
<td>8/13/14</td>
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<td>FINAL MTV AEROBRAKE ASSEMBLY MISSION</td>
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<tr>
<td>MTV TEI, HABITAT MODULE ASSEMBLY AND MTV/MEV ASSEMBLED</td>
<td>11/27/14</td>
<td>3/13/16</td>
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<tr>
<td>TMIS CORE STACK ASSEMBLED TO MTV/MEV ASSEMBLY</td>
<td>3/13/15</td>
<td>6/27/15</td>
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<td>TMIS PROPELLANT TANKS FIRST ASSEMBLY MISSION</td>
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<td>TMIS PROPELLANT TANKS SECOND ASSEMBLY MISSION</td>
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<td>1/28/16</td>
<td>2/3/16</td>
<td>HLLV MISSION ELEVEN</td>
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</tbody>
</table>
On-Orbit Assembly
On-Orbit Assembly
On-Orbit Assembly
On-Orbit Assembly

**MMV TOP ASSEMBLY**

<table>
<thead>
<tr>
<th>Date</th>
<th>Event Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>6/28/13</td>
<td>Initial NEV Aerodynamic Assembly Mission</td>
</tr>
<tr>
<td>7/15/13</td>
<td>MMV Descent, ascent, surface and science payload assembly</td>
</tr>
<tr>
<td>8/12/14</td>
<td>Initial MTV Aerodynamic Assembly Mission</td>
</tr>
<tr>
<td>9/27/14</td>
<td>MTV Tel, habitat module assembly, MMV assembly</td>
</tr>
<tr>
<td>10/13/14</td>
<td>This propellant tanks second assembly mission</td>
</tr>
<tr>
<td>1/20/15</td>
<td>This propellant tanks final assembly mission</td>
</tr>
</tbody>
</table>

- Smallest unit of time is 1 man-hour
- 16 man-hours = 1 man-day of Assembly Duration

**BASELINE DURATIONS:**

- HILV Launch = .5 man-day
- HILV achieves stable orbit = .25 man-day
- OMV deploys from/to Freedom = .5 man-day
- OMV berths to components = .25 man-day
- Unstore and power up Robotics = .06 man-day
- Robotic verification = .12 man-day
- HILV deploys components = .06 man-day
- OMV transfers components = .25 man-day
- Robotic tasks = .06 man-day
- EVA/Robotic Contingency = .5 man-day
- Component Inspection = .12 man-day
- Component Test = .25 man-day

- Subassemblies to stand-by mode = .5 man-day
- Mechanical Fastening of components = .18 man-day
# On-Orbit Assembly

<table>
<thead>
<tr>
<th>Name</th>
<th>Earliest Start</th>
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<td>1/12/14</td>
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<td>MEV DESCENT, ASCENT, SURFACE AND SCIENCE PAYLOAD ASSEMBLY</td>
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<td>106</td>
</tr>
<tr>
<td>MTV TEI, HABITAT MODULE ASSEMBLY AND MTV/MEV ASSEMBLED</td>
<td>11/27/14</td>
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<td>106</td>
</tr>
<tr>
<td>TMIS PROPELLANT TANKS FINAL ASSEMBLY MISSION</td>
<td>1/26/16</td>
<td>2/3/16</td>
<td>HLLV MISSION ELEVEN</td>
<td>8</td>
</tr>
</tbody>
</table>
On-Orbit Assembly

D615-10026-2
HLV MISSION ONE

length of time processing in workdays, based on a 16 hour workday
HILLY MISSION TWO
HLLY MISSION FOUR
MTV ASSEMBLY
HLLY MISSION EIGHT
TMIS PROPELLANT TANKS

- Union and Power up Prim
- Primar Functioning Ability
- Assembly Platform Grapples Propellant Tank with Prim
- PMHS Transfers Propellant Tank to Assembly Position
- Power Umbilical Installed Remotely
- Remote Power up Propellant Tank
- Initial Inspection and Checkout
- EVA/Robotic Contingency Repair
- Power Down and Disconnect Power Umbilical from Prim Propellant Tank

- Can Separates From Core Stack
- Can Returns to Canister Maintenance
- Can Returns to Freedom
- Can Transfers to Canister Deployment Mechanism
- Can Transfers Propellant Tank to Assembly Platform
- Caneither with TMIS Propellant Tank
HLLV MISSION TEN

[Diagram showing a flowchart of HLLV mission steps]

- Tank propellant tanks processed for launch
- HLLV launches to assembly platform
- HLLV achieves stable position at assembly platform
- Can deploy from freedom and rendezvous with HLLV
- HLLV propellant tank assembled to main assembly
- HLLV processed for next mission
H.L.V. MISSION ELEVEN

- This propellant tanks processed for launch
- Kelly launched to Assembly Platform
- Kelly achieves stable position at Assembly Platform
- CM deploys from platform and rendezvous with Kelly
- SME propellant tank assembled to H.L.V.
On-Orbit Support Equipment

- Assembly scenario is to complete major assemblies robotically with crew support for contingency only.

- Robotic operations will be controlled by Ground Control Center primarily, with control being "handed off" to assembly crew during contingency operations.

- Platform Remote Manipulating System (PRMS)
  - Two complete systems required for MMV assembly
  - Each main arm can span the entire 30M diameter of the aerobrake
  - Each main arm has a "hold down" grapple to secure the working end to EVA handrails
  - Each main arm has a 2.5M work arm capable of precise movements and operations
  - Elbow joints feature n-pi rotational freedom and the wrist joints are compact roll-pitch-roll units
  - Video cameras allow direct monitoring and machine vision from the end effector
  - All hardware is bar-coded for positive machine recognition
  - End effector is equipped with a 6-axis EM antennae, which determine location and orientation relative to EM beacons distributed across the assembly site.
  - Tools and hardware required for assembly operations will be secured to the main arm, within reaching distance of the work area
  - Each arm will be capable of maneuvering 128 metric tons (proposed mobile servicing center 10-12-89)
  - Each arm will be track-mounted so as to maneuver about the perimeter of the assembly area
On-Orbit Support Equipment (cont'd)

- Remote Aerobrake Manipulating System (RAMS)
  - Same characteristics as PRMS for commonality
  - 2 systems attached to track on each aerobrake

- Platform Anchor System (PAS)
  - 4 units required for assembly
  - Track-mounted to allow movement during assembly operations
  - Extendable to TBD height to allow access for TPS installation and inspection
  - Lift capabilities of 128 metric tons
  - Grapple-type end effector to anchor components to platform
  - Elbow joints with 0.50pi rotational freedom
  - Wrist joints with roll-pitch-roll movements

- Assembly Support Fixture (ASF)
  - Fixed storage locations. TBD units required for assembly across the assembly platform
  - Removable grapple-type end effector
  - Able to support up to 128 metric tons
  - Grapple fitting remotely controlled to release and secure components

- Lighting & Video Monitoring
  - PRMS and RAMS assembly arms will have required lighting and video/fiber optic monitoring capabilities.
  - Portable lighting will be available as required

- EVA Handrails and Tether Tie-Down Points
  - Available on each assembly component
On-Orbit Support Equipment (cont'd)

- **Electrical Power**
  - Will be supplied by assembly platform (Solar Dynamics)
- **MTV Habitat Module**
  - Provide crew stationing facility
- **SSF Type Node**
  - Houses local control of MAP and Assembly Equipment
  - Provide berthing port for Logistics Module
  - Provide berthing port for ACRV
- **SSF-Type Logistics Module**
  - Provide consumables storage and transportation
Issues and Concerns

- TPS Installation and Inspection
- Launch Vehicle Integration
- Robotic Operations
- Mechanical Fasteners vs. Welding
Revised On-Orbit Analysis

The following two charts summarize the results of the Revised On-Orbit Assembly Analysis. Revisions were made because the analysis for HLLV processing was found to be in series to On-Orbit Assembly, whereas the analysis should have been in parallel.
On-Orbit Assembly Analysis

- Assembly Analysis Presented at Second Quarter Review has Been Revised

- Revised Data shows 5 months less time required for On-Orbit Assembly
  - Original HLLV ground processing time was calculated in series with the On-Orbit Assembly time
  - Revised analysis calculates the HLLV ground processing time in parallel with the On-Orbit Assembly time

- Original On-Orbit Assembly Start date of January 2014

- Original On-Orbit Assembly Completion date of December 2016

- Revised On-Orbit Assembly Start date of August 2013

- Revised On-Orbit Assembly Completion date of February 2016

REVISED ASSEMBLY SCHEDULE
# On-Orbit Assembly Analysis

## Advanced Civil Space Systems

### Boeing

<table>
<thead>
<tr>
<th>Name</th>
<th>Earliest Start</th>
<th>Earliest Finish</th>
<th>Subproject</th>
<th>Days</th>
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<tbody>
<tr>
<td>Habitat Module and Surface Payload Assembly Mission</td>
<td>1/1/13</td>
<td>10/31/13</td>
<td>HILLY Mission One</td>
<td>9</td>
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<tr>
<td>Initial MEV Aerobrake Assembly Mission</td>
<td>10/31/13</td>
<td>1/30/14</td>
<td>HILLY Mission Two</td>
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<tr>
<td>Final MEV Aerobrake Assembly Mission</td>
<td>1/30/14</td>
<td>6/1/14</td>
<td>HILLY Mission Three</td>
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<td>MEV Descent, Ascent, Surface and Science Payload Assembly</td>
<td>6/1/14</td>
<td>7/31/14</td>
<td>HILLY Mission Four</td>
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<tr>
<td>Initial MTV Aerobrake Assembly Mission</td>
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<td>10/30/14</td>
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<td>Final MTV Aerobrake Assembly Mission</td>
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<td>1/29/15</td>
<td>HILLY Mission Six</td>
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<td>MTV Tei, Habitat Module Assembly and MTV/MEV Assembled</td>
<td>1/29/15</td>
<td>4/30/15</td>
<td>HILLY Mission Seven</td>
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<tr>
<td>IMS Core Stack Assembled to MTV/MEV Assembly</td>
<td>4/30/15</td>
<td>7/30/15</td>
<td>HILLY Mission Eight</td>
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<td>IMS Propellant Tanks First Assembly Mission</td>
<td>7/30/15</td>
<td>10/29/15</td>
<td>HILLY Mission Nine</td>
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<td>IMS Propellant Tanks Second Assembly Mission</td>
<td>10/29/15</td>
<td>1/28/16</td>
<td>HILLY Mission Ten</td>
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<td>IMS Propellant Tanks Final Assembly Mission</td>
<td>1/28/16</td>
<td>2/5/15</td>
<td>HILLY Mission Eleven</td>
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</table>

**Diagram:**

- Habitat Module and Surface Payload Assembly Mission
- Initial MEV Aerobrake Assembly Mission
- Final MEV Aerobrake Assembly Mission
- MEV Descent, Ascent, Surface and Science Payload Assembly
- Initial MTV Aerobrake Assembly Mission
- Final MTV Aerobrake Assembly Mission
- MTV Tei, Habitat Module Assembly and MTV/MEV Assembled
- IMS Core Stack Assembled to MTV/MEV Assembly
- IMS Propellant Tanks First Assembly Mission
- IMS Propellant Tanks Second Assembly Mission
- IMS ProPELLANT Tanks Final Assembly Mission
Node Purpose & Minimal Requirements

Purpose

- Integration
- De-integration
- Re-integration

of mission vehicles, including assembly, processing, resupply and refurbishment

Top-level Requirements

- Accessibility
- Provision of support services to mission vehicle

Functional Approach

- Analyze specific functions necessary to provide required services to the mission vehicles
- Identify synergistic ways of providing those functions, emphasizing "operational" solutions (e.g. using/proving onboard vehicle systems, resupplying before mission departure)
- Defer device-driven solutions until minimum common requirements are distilled, which cannot be satisfied by hardware already "procured" for the vehicle itself
Assembly Node Purpose and Requirements

Assembly Node Purpose and Minimal Requirements was developed to support a study that MSFC was performing to determine the advantages and disadvantages of different Orbits for an Assembly Node. Each vehicle option has advantages for being assembled and launched from different Orbits. The data shown in the following charts is for the Cryo/Aerobrake Vehicle.
# Node Comparisons

<table>
<thead>
<tr>
<th>Node</th>
<th>LEO MASE REF.</th>
<th>GEO Case 1</th>
<th>GEO* Case 3</th>
<th>LLO*</th>
<th>L2*</th>
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</thead>
<tbody>
<tr>
<td>Data to Mars</td>
<td></td>
<td></td>
<td></td>
<td></td>
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<tr>
<td>Δ V to Mars (m/sec)</td>
<td>4281</td>
<td>5402</td>
<td>2423</td>
<td>1995</td>
<td>1315</td>
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<tr>
<td>TMI S mass</td>
<td>470.3 t</td>
<td>751.5 t</td>
<td>191.8 t</td>
<td>147.7 t</td>
<td>87.9 t</td>
</tr>
<tr>
<td>Debris Env.(total time)</td>
<td>Continuous</td>
<td>0</td>
<td>&lt; 1 hr</td>
<td>&lt; 1 hr</td>
<td>&lt; 1 hr</td>
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<tr>
<td>Radiation Env. total time in Van Allen belts</td>
<td>~2hrs</td>
<td>0</td>
<td>&lt; 2 hrs</td>
<td>&lt; 2 hrs</td>
<td>&lt; 2 hrs</td>
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<tr>
<td>Radiation Environment</td>
<td>Trapped, SAA</td>
<td>Trapped, GCR, SPE</td>
<td>Trapped, GCR, SPE</td>
<td>GCR, SPE</td>
<td>GCR, SPE</td>
</tr>
<tr>
<td>Crew to Node ΔV /t</td>
<td>SSF***</td>
<td>SSF - 6 hrs</td>
<td>SSF - 6 hrs</td>
<td>SSF - 13 days</td>
<td></td>
</tr>
<tr>
<td></td>
<td>ΔV=10-100m/sec hrs-days</td>
<td>ΔV= 4200m/sec</td>
<td>ΔV= 4200m/sec</td>
<td>ΔV= 4000 m/s Moon - 2hr ΔV= 2100m/s</td>
<td></td>
</tr>
<tr>
<td></td>
<td>e.s.o.** ~ 10 min orbit align ~ 5d opps/day ~15-16 rec = 30-60 days</td>
<td>e.s.o.~ 1/day orbit align ~ planetary position opp ops/day = continous</td>
<td>e.s.o. @ 10 min alignment =12hr rec.= 27 days retrograde opp.</td>
<td>alignment =12hr rec. = 27 days retrograde opp.</td>
<td></td>
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<tr>
<td>Launch window timing</td>
<td></td>
<td></td>
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<td></td>
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<tr>
<td>Logistics window timing</td>
<td>anytime</td>
<td>5 hr transit, 2 opportunities/day</td>
<td>every 10 days</td>
<td>transit from Moon = 3 day</td>
<td></td>
</tr>
</tbody>
</table>

* denotes PEGA (powered Earth gravity assist)  
GCR = Galactic Cosmic Radiation  
** engine start opportunity  
SPE = Solar Proton Events  
*** co- orbiting with SSF costs ΔV to maintain position relative to SSF (not continuous thrust)
## Assembly Node Concepts Pros and Cons

<table>
<thead>
<tr>
<th>Node Concepts</th>
<th>Key Features/Advantages</th>
<th>Key Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dedicated Assembly Node</td>
<td>• Abundant storage&lt;br&gt;• Totally self-contained&lt;br&gt;• Vehicle systems unused&lt;br&gt;• Multiple robot arms&lt;br&gt;• Sections of vehicle may be assembled simultaneously</td>
<td>• Larger than SSF&lt;br&gt;• Will take long time to construct&lt;br&gt;• Excessive reboost requirements&lt;br&gt;• Mechanically complex&lt;br&gt;• Local debris shielding required&lt;br&gt;• Must be in place prior to vehicle assembly</td>
</tr>
<tr>
<td>I-Beam Platform</td>
<td>• Can be carried up in first HLLV flight&lt;br&gt;• Can easily reach most parts of vehicle with two robot arms&lt;br&gt;• Uses vehicle for comm., data, RCS, power after initial deployment&lt;br&gt;• Can serve as base for experiments</td>
<td>• Fuel cells, batteries required for initial deployment&lt;br&gt;• Limited storage area&lt;br&gt;• Precursor mission required for deployment</td>
</tr>
<tr>
<td>&quot;Smart&quot; HLLV Platform</td>
<td>• No additional platform required&lt;br&gt;• HLLV shroud provides limited debris shielding&lt;br&gt;• HLLV provides for communication, data, RCS, GNC, etc.&lt;br&gt;• Robot arms transferable to NTR</td>
<td>• Increased HLLV complexity&lt;br&gt;• Reboost fuel has to be replenished&lt;br&gt;• Limited storage&lt;br&gt;• Vehicle must be detached from HLLV prior to assembly complete&lt;br&gt;• Local debris shielding required</td>
</tr>
<tr>
<td>Hinged Truss Platform</td>
<td>• Uses vehicle truss as assembly platform; no other platform needed&lt;br&gt;• Reach to remote engine section of vehicle provided by flexing truss at hinges&lt;br&gt;• Vehicle subsystems used; no additional systems necessary</td>
<td>• Requires a precursor mission to deploy truss&lt;br&gt;• Batteries, fuel cells necessary for initial deployment&lt;br&gt;• Reboost, comm., data, power, must be in place prior to assembly start&lt;br&gt;• Limited storage&lt;br&gt;• Local debris shielding required</td>
</tr>
<tr>
<td>Vehicle as its own Platform</td>
<td>• Reduces needed on-orbit infrastructure&lt;br&gt;• Deletes additional facilities and resources needed for designing, building, launching, and maintaining separate assembly platform</td>
<td>• Requires dedicated HLLV flight for non-optimized packaged first element&lt;br&gt;• Requires vehicle to have additional control, reboost&lt;br&gt;• No additional storage&lt;br&gt;• Requires batteries or fuel cells for initial deployment&lt;br&gt;• Requires localized debris shielding</td>
</tr>
<tr>
<td>Node Concepts</td>
<td>Key Features/Advantages</td>
<td>Key Disadvantages</td>
</tr>
<tr>
<td>-------------------------------</td>
<td>----------------------------------------------------------------------------------------</td>
<td>-----------------------------------------------------------------------------------</td>
</tr>
<tr>
<td>Assembly Flyer Platform</td>
<td>• Performs HLLV unloading, payload/crew transport, and assembly with one vehicle</td>
<td>• No additional storage</td>
</tr>
<tr>
<td></td>
<td>• Compatible with SSF</td>
<td>• Requires vehicle to have additional control and reboost systems</td>
</tr>
<tr>
<td></td>
<td>• Capable of manned/robotic operations</td>
<td>• Requires development and production of sophisticated man-rated space vehicle</td>
</tr>
<tr>
<td></td>
<td>• Uses CTV for main P/A</td>
<td>• Requires localized debris shielding</td>
</tr>
<tr>
<td></td>
<td>• Can serve as free flying platform between assemblies</td>
<td></td>
</tr>
<tr>
<td>SSF Based Assembly of First</td>
<td>• Uses planned SSF growth concept</td>
<td>• Impact to SSF (resources, microgravity, drag, etc.)</td>
</tr>
<tr>
<td>Element</td>
<td>• Provides quick and easy crew logistics access to initial assembly operations</td>
<td>• Eventually requires vehicle to have additional control and reboost systems</td>
</tr>
<tr>
<td></td>
<td>• Allows verification and checkout of critical systems prior to independent vehicle operations</td>
<td>• Requires localized debris shielding</td>
</tr>
<tr>
<td></td>
<td>• Does not disrupt SSF operations beyond first assembly mission (remainder of assembly based from vehicle itself)</td>
<td>• No additional storage beyond first element</td>
</tr>
<tr>
<td>Tethered off-SSF Assembly</td>
<td>• Compatible with current SSF design</td>
<td>• Impact to SSF resources</td>
</tr>
<tr>
<td>Platform</td>
<td>• Provides quick and easy crew and logistics access to entire assembly and propellant transfer operations</td>
<td>• Requires localized debris shielding</td>
</tr>
<tr>
<td></td>
<td>• Microgravity and dynamic loads impacts to SSF minimized by tether</td>
<td>• No additional storage</td>
</tr>
<tr>
<td></td>
<td>• Removes hazardous operations and materials to SSF standoff distance</td>
<td>• Requires additional reboost and control systems on SSF</td>
</tr>
</tbody>
</table>
Ground
Integrated Aerobrake Launch Option

On-Orbit Assembly of the Mars Aerobrake(s) require two 10.5 meter dia HLLV launches each and 180 days (due to 90 day ground processing time required for the HLLV). A concept which would deliver an assembled aerobrake or aerobrakes to LEO is shown in the following chart. This concept utilizes a Shuttle derived In-Line vehicle to launch two aerobrakes to LEO.
Shuttle Derived Aerobrake Launch Option

ADVANCED CIVIL SPACE SYSTEMS

- Aerobrake completely assembled on Earth.
- Eliminates in-orbit assembly/check-out.
- Eliminates packing problem.
- Uses shuttle derived launch vehicle.
- Can possibly pack MBV lander structure and descent tanks in aerobrake at launch.
- Recommended launch vehicle for Lunar/Mars Initiative.
- 90t launch capacity.

Modified B.T.

Standard SRB's

(2) Aerobrakes

(3) SSMB's

Aerobrake Mass
MEV 15,784 kg
MTV 24,650 kg
Integrated Aerobrake Launch Option

The following chart shows a Shuttle-C / Aerobrake Integrated Launch Option. This option would launch a single Aerobrake to LEO along with other payload stored in the Shuttle-C Payload Shroud.
Size Comparison SSF Launch Facility With Aerobrake Footprint

This is a diagram of the low L/D aerobrake with its requiremented access corridor, space for general support equipment (GSE) and assembly activities superimposed on the footprint of a space station payload, and over four adjoining SSF payload footprints. The Space Station facility working area is 5 of the SSF payload footprints, it will be to small for aerobrake assembly and manipulation. A new facility will be needed.
Size Comparison SST Launch Facility with Aerobrake Footprint

Access - 45' x 90' (Envelope)
Access Equipment, GSE 35' x 75' (Envelope)
Space Station Launch Package 15' x 60'
Aerobrake 100' x 90'
Assembly 110' x 100'
GSE 125' x 115'
Access 135' x 125'

Space Station Launch Facility
Mars Aerobrake Assembly and Integration  
at the VAB

The current Vehicle Assembly Building (VAB) has a transfer aisle with the dimensions of 92.5 feet* the aerobrake is 91.8 feet,* as shown in the following chart, the Aerobrake can be transferred through the transfer aisle horizontally.

* 28.2 meters
• 28 meters
Mars Aerobrake Assembly at VAB

ADVANCED CIVIL SPACE SYSTEMS

BOEING
Mars Aerobrake Launch Option

Modifications of existing equipment will be required to assemble and check out the Integrated Launch Option in the current VAB High Bay.
Launch Site Impacts

The following Launch Site Impacts were derived from current facility and equipment limitations. Refurbishment and modification of existing equipment or construction of new facilities will accommodate an Integrated Launch Option.
Launch Vehicle / Integral Aerobrake
Launch Site Impacts

- Transporter required for fully assembled Aerobrake
- VAB High Bay access platforms will require modifications
- Aerobrake will prohibit use of the Rotating Service Structure without major modifications
- Fixed Service Structure swing arm extension and retraction may interfer with the Aerobrake
- Large Aerobrake cross-sectional area will impart large wind loads to the launch vehicle
  - Increased loads to hold down fixtures
  - Revised launch commit criteria for maximum winds at launch
Aerobrake Preflight Operations

We have began analyzing the necessary functions that must be performed to the aerobrake, the largest, most fragile and difficult to launch, piece of spacecraft. This chart and the following one give the analyses of functional flows for ground operation and requirements.
Ground Assembly and Check Out

The following chart shows the Ground Rules and Assumptions developed for Ground Assembly Analysis of the Cryo/Aerobrake Vehicle.
Ground Rules / Assumptions
For Ground Assembly

ADVANCED CIVIL SPACE SYSTEMS — BOEING

- A System is a group of components and supporting structure that is integrated by a contractor and delivered as a unit to the processing facility (e.g. MEV Aerobrake, MEV Descent Lander, Ascent System, etc.).

- System Interfaces are those which transmit data, power, or fluids across the system's boundaries and mechanically secure one system to another.

- Subsystems Interfaces are those which are internal to a System.

- Subsystem Interfaces are verified by the manufacturer prior to System integration.

- Component Interfaces are those which are internal to a Subsystem.

- Component Interfaces are verified by the manufacturer during Subsystem Assembly.

- Interfaces verified prior to System Level Integration will be accepted with no repetition of tests.

- Flight Hardware will be used to verify System Interfaces.

- Ground facilities will simulate assembly node operations and limitations.
MMV Top Level System Interfaces

The following chart shows the Top Level System Interfaces of the Cryo/Aerobrake Vehicle.
MMV System Interfaces

ADVANCED CIVIL SPACE SYSTEMS  BOEING

Descent Lander System

Payloads Science and Surface

Ascent System

Trans Earth Injection System

MEV Aerobrake

TMIS Assembly

MTV Crew Transfer Modules

MTV Aerobrake
MMV System Interfaces

The following two charts show Cryo/Aerobrake System and Subsystem Interfaces. The interfaces shown are major interfaces, that is, one electrical interface may represent several electrical cables. The total number of component level interfaces has not been defined at this point.
MMV System Interfaces

Legend:

- M Mechanical Interface
- E Electrical Interface
- D Data Interface
- Fluid Interface

System Level
- M Mechanical Interface
- E Electrical Interface
- D Data Interface
- Fluid Interface

Subsystem Level
Assembly Node Interfaces

The following chart shows the Assembly Node interfaces to Cryo/Aerobrake. The Assembly Node requirements and equipment interfaces were developed by the On-Orbit assembly analysis.
Assembly Node Interfaces

Assembly Node

Legend

Subsystem Level
- M — Mechanical Interface
- D — Data Interface
- E — Electrical Interface
- F — Fluid Interface
- L — Life Support System Interface
- C/D — Control/Data Interface

System Level
- M —
- D —
- E —
- F —
- L —
- C/D —

MTV Aerobrake
MBV Aerobrake

MTV Habitat Module

Assured Crew Rescue Vehicle

Resource Node (If Required)

Platform Remote Manipulating System
Platform Power Source
Assembly Support Fixture
Platform Anchor System
Pressurized Logistics Module

565

TCABEN/viz/206xrsz50
The following chart defines the process of Sequential Interface Verification for Ground Processing of the Cryo/Aerobrake Vehicle.
Sequential Interface Verification

ADVANCED CIVIL SPACE SYSTEMS

• Process of verifying the interfaces of the Mars Mission Vehicles elements without complete assembly.

• Elements are received and inspected at the assembly area.

• Internal test performed and certified by the contractor will not be repeated.

• Elements will be assembled to the level required to verify the interfaces from one element to another.

• Interfaces will be verified by flight hardware when feasible or by mate devices/prototypes when necessary.

• Elements will be disassembled to payload configurations and processed for launch.
Ground Processing Functional Flow

The following three charts show the Functional Flow of Ground Processing for the Cryo/Aerobrake Vehicle. This Flow is a top level flow that shows the requirements for sequential interface verification.
Ground Processing Functional Flow

1. Receiving and Inspection of MTV Habitat Module
   - Assemble MTV Habitat Module to TEI
   - Verify Interfaces of MTV Habitat Module and TEI
   - Disassemble TEI and MTV Habitat Module
   - Process TEI for Launch

2. Unstore TMIS Truss Structure and Receive and Inspect TMIS Tanks
   - Assemble TMIS Truss Structure and Tanks
   - Verify Interfaces of Truss Structure and Tanks
   - Disassemble TMIS and MTV Habitat Module
   - Process MTV Habitat Module for Launch
   - Process TMIS for Launch

3. Assemble TMIS to MTV Habitat Module
   - Verify Interfaces of TMIS and MTV Habitat Module
   - Disassemble TMIS
Cryo/Aerobrake Test Philosophy

The following chart is the summary of the Test Philosophy developed for this analysis. The complete Philosophy and Approach was included in the May Progress Report.
Test Philosophy

Purpose: Establish criteria and overall test approach that verifies a system is flight ready and will accomplish its mission successfully

Goals:
- Reduce redundant testing
- Reduce man power requirements for ground processing
- Reduce overall cost
- Provide system operational history

Criteria:
- Self test software and peripheral equipment will perform mechanical, electrical and electronic system tests and readiness analysis in an autonomous fashion
- Redundant flight hardware (on-board systems) will be under continuous self check
- Physical interfaces will be self latching connectors
- Prototype systems will be utilized when feasible for ground processing activities
- Commonality of systems will be stressed
Ground Processing Facility Requirements

The following chart is a preliminary analysis of the Facility Requirements for Ground Processing of the Cryo/Aerobrake Vehicle.
Facility Requirements

ADVANCED CIVIL SPACE SYSTEMS

- Processing Facility Ground Rules:
  Utilize Standard Services: Cranes, Power, Communications, Clean Rooms, etc.
  Make Unique Hardware Portable: Special Test Equipment, Work Stations, Handling Fixtures, etc.
  Provide Large Volume Workspace that can be readily adapted to System Block changes,
  Multiple Systems in Flow.
  Provide for Hazardous System Processing

- Equipment Requirements
  Overhead Crane
  Flat Floor / Air Pallets
  Standard Commercial Power
  Uninterrupted Instrumentation Power
  Environmental Control System: Humidity 50 +/- 5%  Temperature 75 +/- 5°F
  100K Cleanliness Level
  Closed Circuit Television
  Facility GN2
  Helium Supply
  Shop Air
  Fire Protection / Deluge
  Shower / Eye Wash
  Vacuum
  Lightning Protection
  Potable Water
  Paging
  Commercial Telephone
  RF System
  Operational Intercom System
  Personnel Airlock
  Grounding
  Transportation/Ground Handling Fixtures
VI. Implementation Plan
Technology Needs and Advanced Plans
Technology Issues - Cryogenic / Aerobraked Vehicle

I. Introduction

Technology issues relating to the reference vehicle are presented in this section. Some of the charts are also included in the NTR, NEP, and SEP IP&ED documents. The focus of this section will be to bring out those issues important to the reference cryogenic vehicle from these charts, and to present a series of technology level requirements necessary for the reference vehicle. The most important technology development needs for this option are in the areas of high energy aerobraking, and cryogenic fluid storage and management.

II. Technology commonality Issues

The following nine charts lay out the important technology commonality issues between the major propulsion options as well as across the seven major mission architectures identified in this study. The reference vehicle exhibits commonality, and therefore is a good “building block” for the other vehicles in several important areas. The transfer crew module is substantially the same as for all the other options. The MEV is identical across all vehicle options, except for the cryogenic propellant management and storage issues. The demands placed on the avionics system for the chemical system are similar to those for the NTR, and probably greater than those needed for the low thrust NEP and SEP options. Finally, in-space assembly issues should be similar for the reference and NTR vehicle, with the exception of the related nuclear issues associated with the NTR. Assembly issues relating to the NEP and SEP, while duplicate in some areas, will be unique in most areas.

The seven identified Lunar/Mars mission architectures versus the required component technologies, enabling and enhancing, are shown on the next set of charts and facing page text. Many of these component technology issues are common across the listed architectures. These issues are for the entire integrated architectures, and do not necessarily refer specifically to the reference vehicle. Cryogenic/aerobraked vehicles are used in most of the architectures for initial Mars missions, and for all early Lunar missions. The areas of high thrust cryogenic propulsion, and high energy aerobraking are the primary areas of technology development concern for the reference option.

III. Technology Development Concerns

As noted before, many of the identified critical and high leverage technology development issues are common across all four major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H₂, and possibly O₂ for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique cryo/AB technology issues include high energy aerobraking, and large advanced space engine advanced development. Enhancing technologies include cryogenic refrigeration (lander tanks), O₂-H₂ RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.
IV. Cryo/Aerobraked Vehicle Technology Requirements

Technology performance levels required for the reference cryogenic vehicle are outlined in the next eight charts. These are not intended to be the levels needed for a minimum vehicle, but serve mainly to document the levels required to accomplish the identified reference mission profile with the vehicle model as configured. Changes to these specifications would not necessarily affect the feasibility of a chemical Mars mission, but would change the reference vehicle configuration. The list also includes operational requirements which could drive technology development or advanced development. An example of this could be the requirement of wet launched tanks, rather than filling on orbit, which would affect tank design, and possibly in-space thermal performance.

V. Cryo/Aerobraked Technology Development Schedule

The final chart in this section is a proposed technology development schedule for the nuclear electric propulsion option. The schedule shows that, given a FY '91 start, the SEP vehicle could be ready for a Mars mission in the 2009 timeframe. A full scale decision point is also highlighted during year 7. This is the point where a commitment should be made for full scale funding and development of the program.
# Technology Commonality and Differences

<table>
<thead>
<tr>
<th>System/Subsystem</th>
<th>Reference (Cryo-A/B)</th>
<th>NTR Vehicle</th>
<th>NEP Vehicle</th>
<th>SEP Vehicle</th>
</tr>
</thead>
<tbody>
<tr>
<td>Crew Systems/Habitats</td>
<td>Deployed solar array system; low power (~50-75 kW). Low temp heat rejection (~400°K)</td>
<td>Common to reference vehicle system</td>
<td>Nuc./Rankine or Brayton cycle energy conv. sys. Very high power level (up to 200 MW). High temp heat rejection (~1000°K-main cycle).</td>
<td>Solar-electric energy conversion. High power (~10 MW or greater) level. Moderate temperature radiators (400 - 650 K).</td>
</tr>
<tr>
<td>Life Support, rad. prot., hab. struct., &amp; airlock/EVA</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Power System &amp; Thermal Control</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Propellant Management &amp; Storage</td>
<td>Long term storage of H2 &amp; O2 for Earth &amp; Mars orbit, and deep space environ. necessary with minimal boiloff. Low-g fluid gaging, acquisition, and transfer highly enhancing or enabling for all missions. NTR requires common techniques for LH2 fuel.</td>
<td>Argon propellant management system can be similar to LOX storage system, but without the safety constraints associated with an oxidizer.</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Propulsion System</td>
<td>Advanced cryogenic space engines with &gt;475 sec Isp, and ~30 kib to ~200 kib thrust.</td>
<td>NERVA derived/advanced NTR system with higher Isp (up to 1050 sec vs. 850 sec.)</td>
<td>Rankine or Brayton cycle conversion system driving cluster of ion thrusters for NEP. Same thrusters for SEP. Number of thrusters depends on available thruster size and required redundancy.</td>
<td></td>
</tr>
<tr>
<td>Aerobraking</td>
<td>Low L/D - AFE derived for Earth capture.</td>
<td>Not needed for Lunar NTR (propulsive capture @ Earth)</td>
<td>Not needed for NEP or SEP.</td>
<td></td>
</tr>
<tr>
<td>Lunar Mars</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Avionics</td>
<td>Avionics system hardware may be common for Lunar or Mars (or L/M growth)</td>
<td>Avionics system required for low &amp; continuous thrust vehicles are lower than for Cryo A/B or NTR vehicle.</td>
<td></td>
<td>Severe LEO debris environ. damaging to solar arrays. Spare set of arrays may be necessary. MEV A/B launch &amp; assembly needed.</td>
</tr>
<tr>
<td>Assembly &amp; Checkout</td>
<td>Common assembly facility &amp; equip. for most mission vehicles. Assembly time in LEO, and thus M/D protection level is varied. Mars vehicle requires launch &amp; assembly of large (~30 m vs. 20 m for Lunar) aeroshell. Nuclear vehicles (NTR &amp; NEP) may face political constraints on launch &amp; assembly of vehicle. Assembly &amp; operation may be necessary from nuclear safe orbit.</td>
<td></td>
<td></td>
<td>_____________________________________________________________________________________________________________________________________________________</td>
</tr>
</tbody>
</table>
A set of required technologies for the seven identified alternative mission architectures outlined in the evolutionary concepts section is presented. The purpose of this matrix is to provide a preliminary comparison of technology development needs for the alternative architectures. The matrix also serves to better define the architectures. From this top level matrix, a more detailed set of technology requirements can be derived. A set of accommodating technologies can be compiled for needs areas where options exist. Finally, the technology areas can be prioritized as enabling and enhancing, and a return on investment performed for identified high leverage technologies. This portion of the matrix includes most of the cryogenic management issues. Enabling technologies are represented by the filled circle, and enhancing technologies by the open circle. Extensive low - g cryogenic propellant launch, acquisition and transfer refers to the Mars conjunction case, and the mass driver option, where propellant will be used for the transfer vehicles, which will be parked in a low - g environment (Lunar or Mars orbit, or libration staging point). The Mars cycler orbit case includes a question mark for the long term cryogenic storage system, because the necessary thrust levels and type of propulsion system are undetermined at this time.
### Required Technologies vs. Alternative Mission Architecture

<table>
<thead>
<tr>
<th></th>
<th>Low boiloff cryogenic propellant storage system (1-3 yrs)</th>
<th>Low boiloff cryogenic propellant storage system (15-60 yrs)</th>
<th>Low - g fluid acquisition and transfer</th>
<th>Extensive low - g cryogenic propellant launch, acquisition, and transfer</th>
<th>Cryogenic tank integrity monitor</th>
<th>Cryo fluid reusability umbilical</th>
<th>Lunar LOX production, liquefaction, and transfer technology</th>
<th>Mars O2 production, liquefaction, and transfer technology</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mars SEP Alternative Architecture</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
</tr>
<tr>
<td>Lunar/Mars NTR Alternative Architecture</td>
<td>🟦</td>
<td>🟦</td>
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<td>🟦</td>
<td>🟦</td>
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<td>🟦</td>
</tr>
<tr>
<td>L2 Node / Mass Driver Alternative Architecture</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
<td>🟦</td>
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</tr>
<tr>
<td>Mars Cycler Orbits Alternative Architecture</td>
<td>🟦</td>
<td>🟦</td>
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<td>🟦</td>
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<tr>
<td>Mars Conjunction/Direct Alternative Architecture</td>
<td>🟦</td>
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</tr>
<tr>
<td>Lunar / Mars NEP Alternative Architecture</td>
<td>🟦</td>
<td>🟦</td>
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</tbody>
</table>

- 🟦 - Enabling
- 🟦 - Enhancing
This matrix section represents the major aerobraking concerns. The aerobraking energy columns for Mars and Earth capture digresses from the format in order to illustrate the energy levels, and therefore, the level of technology development needed for the various architectures. Aeroheating predictions, reusable aerobrake TPS, advanced GN&C, and TT&C follow along with the high and medium energy missions. Again, a question mark is shown for the Mars cycler orbit case. Reusable TPS for Earth return cannot be determined as a technology development concern until the aeroheating load at Mars can be determined for the cycler orbits. Further mission design efforts must be carried out before an estimate on this can be made.
### Required Technologies vs. Alternative Mission Architecture (Cont.)

<table>
<thead>
<tr>
<th>Alternative Architecture</th>
<th>Earth return aerobrake energy</th>
<th>Mars capture aerobrake energy</th>
<th>Mars lander aerobrake</th>
<th>High performance aerobrake structure</th>
<th>Aerobrake assembly and test</th>
<th>Aeroheating prediction (Earth and/or Mars)</th>
<th>Reusable aerobrake TPS for Earth return</th>
<th>GN &amp; C to protect TPS</th>
<th>Advanced high accuracy and rate TT &amp; C</th>
<th>In space AR&amp;D/assembly</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mars NEP</td>
<td>Low</td>
<td>Low</td>
<td>●</td>
<td>●</td>
<td>●</td>
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<tr>
<td>Lunar/Mars NTR</td>
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<tr>
<td>Mars SEP</td>
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<td>Low</td>
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<tr>
<td>L2 Node / Mass Driver</td>
<td>High</td>
<td>High</td>
<td>●</td>
<td>●</td>
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<td>●</td>
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</tr>
<tr>
<td>Mars Cycler Orbits</td>
<td>High</td>
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<td>●</td>
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<tr>
<td>Mars Conjunction/Direct</td>
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<tr>
<td>Lunar / Mars NEP</td>
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</tbody>
</table>

- **Enabling**
- **Enhancing**
This matrix area represents the major propulsion issues, with the exception of the radiation protection system, for the baseline and alternative mission architectures. This system, which uses the inert and can waste for radiation shielding, can be enhancing, while a GCR and ALSPE shelter is enabling for all mission architectures. Again, due to the undefined MARS cycler orbit trajectories, it is questionable as to the need for a large cryogenic space engine. A H2-O2 ACS/RCS system is noted as enabling for each option, as it will be for any option over a baseline storable system. A lunar orbital momentum storage and transfer device such as a bolo can be enhancing for all missions, after an initial launch and assembly penalty for the massive (~1000 t) device.
### Required Technologies vs. Alternative Mission Architecture (Cont.)

<table>
<thead>
<tr>
<th>Alternative Architecture</th>
<th>Large (150-200 kib) cryogenic advanced space engine</th>
<th>Small (15-30 kib) cryogenic advanced space engine</th>
<th>H2-O2 ACS/RCS</th>
<th>Multi-MW space based nuclear electric power</th>
<th>Multi-MW space based nuclear thermal power</th>
<th>Surface nuclear electric power</th>
<th>Multi MW solar power system (arrays and handling equip.)</th>
<th>Radiation protection (system to inert &amp; can waste)</th>
<th>Mass driver / rail gun technology</th>
<th>Lunar orbital momentum transfer device (Bolo)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mars NEP</td>
<td>●</td>
<td>●</td>
<td>●</td>
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<tr>
<td>Lunar/Mars NTR</td>
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<tr>
<td>Mars SEP</td>
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<tr>
<td>L2 Node / Mass Driver</td>
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<tr>
<td>Mars Cycler Orbits</td>
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<tr>
<td>Mars Conjunction/Direct</td>
<td>●</td>
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<td>●</td>
</tr>
<tr>
<td>Lunar / Mars NEP</td>
<td>●</td>
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</tbody>
</table>

- **●** - Enabling
- **○** - Enhancing
Required Technologies vs. Alternative Mission Architecture (Cont.)

The final section of the matrix is not as illustrative as the others, in that all of the listed technologies are enabling, with the exception of a closed ecological life support system, which is significantly enhancing for all identified mission architectures.
# Required Technologies vs. Alternative Mission Architecture (Cont.)

<table>
<thead>
<tr>
<th></th>
<th>Autonomous health monitoring and check-out</th>
<th>High data rate comm. or high performance compression</th>
<th>DMS/system diagnostics, Art. intell/neural net/high processing rate GN&amp;C</th>
<th>Long duration refurbishable crew habitat</th>
<th>Long duration BCLSS</th>
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Mars Reference Vehicle Technology Requirements

ADVANCED CIVIL SPACE SYSTEMS

I. TMIS
   A. Cryogenic storage system
      1. Thermal protection system - MLI over foam. (1" foam; ~ 1" MLI)
      2. Tanks launched wet.
      3. Thermodynamic vent coupled to a single vapor cooled shield.
      4. Topoff before Earth departure.
      5. ~ 6 months in LEO before use.
   
   B. Propulsion
      1. Isp = 475 s
      2. Thrust = 150 klb/engine
      3. Advanced space engine.
      4. Nozzle area ratio = 400
      5. No throttling requirements.
      6. Gimbal angle (nominal) = 10°
      7. Up to 3 burns for departure maneuver (2 restarts).
      8. Engine out capability (crossfeed propellant lines).
     10. In-space changeout capability.
     11. Off vehicle preflight checks.
     12. No retraction / extension required.

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Mars Reference Vehicle Technology Requirements (cont.)

C. Structure
   1. Material - metal matrix composites, advanced alloys, and organic matrix composites.
   2. Meteor/debris protection provided for tanks and plumbing.

D. Avionics
   Piggybacked on MTV.

E. Power
   1. Level: < 1 kW
   2. System: Auxiliary power units on engine pod; piggybacked on MTV for back-up.

F. Assembly
   1. Off station assembly.
   2. Degree of assembly: Separate tanksets / propulsion modules connected in LEO to form propulsion stage.

II. MTV

A. Cryogenic storage system
   1. Thermal protection system - ML; 100 layers on H2 & O2 tanks (2").
   2. Tanks launched wet - no transfer other than topoff before Earth departure.
   3. Thermodynamic vent coupled to a series of vapor cooled shields on the H2 tank, and one on the O2 tank.
   4. Topoff in LEO before Earth departure.
   5. ~9 months in LEO before Earth departure.
   6. Boiloff loss of < 10% before Mars departure.
B. Propulsion
1. Isp = 475 s.
2. Thrust = 30 klb/engine.
3. Nozzle area ratio = 400.
4. No throttling requirements.
5. Gimbal angle (nominal) = 10°
7. 3 burns @ 4 - 6 month intervals - minimal degradation.
8. 2 restart capability.
9. Engine out capability (crossfeed propellant lines).
10. Expander cycle.
11. In-space change out capability.
12. Off vehicle preflight checks.
13. No retraction/extension required.

C. Structure
1. Vehicle
   b. Micrometeoroid protection for habitat structure (shell and insulation).
Mars Reference Vehicle Technology Requirements (cont.)

2. Aerobrake
   a. L/D = 0.5
   b. Crossrange: NA
   c. Vhp = 7.07 km/s.
   d. Max-g loading = 6.
   e. Max. temperature = 4000° F.
   f. Structure material: Carbon Magnesium ribs ($\sigma_{ult} = 200$ ksi) bonded to titanium honeycomb shell.
   g. TPS material: Advanced reradiative tiles.
   h. Relative wind angle (reference) = 20°.

D. Avionics
   1. Planetary vicinity -
      a. Relative velocity error = 100 m/s.
      b. Relative position error = 25 km.

   2. System -
      a. Relative velocity error = 100 m/s.
      b. Relative angle error = 0.5°.

E. Power
   1. Level - 15 kW.
   2. System: Solar arrays with battery storage (NiCad).
   3. Back up system: NA
F. Assembly
   1. Off station assembly.
   2. Assembly level (complexity): TBD

G. Habitat
   1. ECLSS: Space Station Freedom derived system with similar degree of closure;
      potable H₂O from cabin condensate; CO₂ reduction/regeneration;
      Hygiene H₂O from urine processing. CELSS to be evaluated.

   2. Structure
      a. 2219 - T8 aluminum pressure vessel.
      b. Pressurized to 20 psig on launch for structural integrity.
      c. Insulation & M/D shield external to pressure shell.
      d. No penetrations in end domes.
      e. Radiation storm shelter provided, and configured to utilize equipment &
         supplies as partial shielding.
      f. External space radiator integral with M/D shield.

   3. Cabin repressurizations: 2+ (outbound emergency could use propellant for
      repress.)
   4. Spares: 15% of active equipment - component level.
   5. Redundancy: Two complete and separate systems for life critical systems +
      spares. Component changeout capability.
   6. Residence time = 535 days.
   7. Science: Transit science as allowed by individual mission.
   8. EVA capability: EVA suits provided for all crew; EVA waste fluid recovery
      for ECLSS.
H. ECCV
1. Apollo size & style as a starting point
2. Open ECLSS (LiOH, no H2O recovery).
3. Residence time: 2 - 3 days.
4. Propulsion: RCS only.

III. MEV
A. Cryogenic storage system
1. Thermal protection system: 100 layers of MLI for H2 and O2 tanks (2").
2. Tanks: double wall tanks with vacuum annulus;
   low thermal conductivity support system for inner tank.
3. Thermodynamic vent: Simple design for gravity field.
4. Tanks launched dry and filled prior to descent, from MTV tanks, or refrigerated. (no boiloff prior to descent)
5. Stay time from 30 - 600 days on Mars surface.
6. Boiloff level < 20% for surface stay.

B. Propulsion
1. Isp = 460 sec.
2. Thrust = 30 klb / engine.
Mars Reference Vehicle Technology
Requirements (cont.)

B. Propulsion (cont.)
   7. No restart capability necessary for nominal case.
   8. Space storage time between burns: NA.
   9. Engine out capability (crossfeed propellant lines).
  10. Expander cycle.
  11. In-space changeout capability.
  12. Off vehicle preflight checks.

C. Structure
   1. Vehicle
      a. metal matrix composites / advanced alloys / organic matrix
         composites.
      b. Micrometeoroid protection for tanks and plumbing.

   2. Aerobrake
      a. L/D = 0.5 to 1.0
      b. Crossrange: 1000 km.
      c. Vhp = 7.07 km/sec.
      d. Maximum g loading: 6.
      e. Maximum temp: TBD (estimated 3100° F).
      f. Structure material: Carbon Magnesium ribs (σult = 200 ksi) bonded
to titanium honeycomb shell.
      g. TPS material: Advanced reradiative tiles.
      h. Relative wind angle (reference) = 20°.
Mars Reference Vehicle Technology
Requirements (cont.)

ADVANCED CIVIL SPACE SYSTEMS

D. Avionics
1. Error without beacon = 1 km.
2. Touchdown error = 1 m/s.
3. Obstacle avoidance capability.

E. Power
1. Level: ~ 2.5 kW.

F. Assembly
1. Off station assembly.
2. Assembly level (complexity): TBD

G. Habitat
1. ECLSS: open system; stored potable H2O; LiOH CO2 adsorption.
2. Structure
   a. Aluminum (2219 - T8) pressure vessel.
   b. Overpressurized on launch for structural integrity.
   c. Insulation and micrometeoroid protection external to pressure vessel.
   d. No penetrations in end domes.
   e. No radiation shelter provided in MEV.
   f. External space radiator integral with micrometeoroid shield.
4. Spares: 15% of active equipment mass; component level.
5. Redundancy: EVA suits as backup to cabin repressurization.; no system level ECLSS redundancy required due to low complexity open system.
6. Residence time: ~3 days (surface systems support surface stay).
7. Science: none.
8. EVA capability: provided for all crew; transferred from MTV.
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Critical Lunar/Mars Reference Technology
Development Concerns

A preliminary set of critical technology development concerns was constructed for the Lunar/Mars reference missions. Its purpose is to show a top level representation of the areas which could prove enabling for the reference Lunar and/or Mars missions, without further concentrated research and development, flight testing, and/or precursor missions. Aerobraking may prove enabling for most Lunar and Mars missions, and significantly enhancing for the rest, primarily due to reduced demands on limited Earth to orbit launch capability and lower launch costs. Aeroheating prediction codes cannot be validated without further experimental data (flight or ground simulation data). The degree of development needed for aerobrake TPS materials will be determined by these predictions. Low gravity human factors, to be evaluated on SSF, may affect vehicle design significantly. For example, vehicle designs must accommodate artificial - gravity until a need level can be determined from space station based research. Finally, precise mission design, incorporating advanced tracking, telemetry, and GN&C must be verified to accommodate aerobraking and automated rendezvous & docking requirements.
A preliminary set of high leverage technologies was assembled for the Lunar/Mars reference missions. These technologies are enhancing for most, and in some cases, all identified mission architectures. Aerobraking will be significantly enhancing for all Lunar and Mars missions where it is not identified as enabling. Other aerobraking issues which could prove enhancing are lightweight reradiative or ablative TPS material, and ECCV vs. aerocapture of MTV at Earth. Low-g propellant handling and low boiloff cryogenic storage are also very enhancing for any missions where it is not enabling. Advanced propulsion options such as NTR, GCR, SEP, and NEP may prove to be high leverage technology options to baseline cryogenic propulsion systems. Finally, developments in advanced materials can be significantly enhancing in a variety of areas.
## Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

<table>
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<tr>
<th>Technology</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aerobraking - Mars Capture (vs. propulsive cap.)</td>
<td>- Aerocapture at Mars can reduce IMLEO &gt;50% over propulsive capture</td>
</tr>
<tr>
<td>Aerobraking - Earth Capture (vs. ECCV)</td>
<td>- ECCV reduces IMLEO and thermal protection system (TPS) requirements.</td>
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<td>- Reusable MTV can reduce life cycle cost.</td>
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<tr>
<td>Aeroshell TPS (reradiative vs. ablative)</td>
<td>- Reusable aeroshell requires rerad. TPS at Mars (or thick lightweight ablator), and ablative at Earth.</td>
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<td>- Further materials and processes advances or low energy mission may allow Earth/Mars reradiative TPS.</td>
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<tr>
<td>Advanced Long Term Cryogenic Storage Technology</td>
<td>- Cryogenic boiloff reduction technologies such as advanced MLI design and application, VCS, para to ortho H2 conv., and thermal disconnect struts, can reduce IMLEO significantly with low R &amp; D effort</td>
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<tr>
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<td>- Longer missions offer greater IMLEO savings potential</td>
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<tr>
<td>Low - g Propellant Transfer</td>
<td>- Low - g propellant transfer technology enhancing for all Lunar/Mars mission arch., and enabling for some Lunar missions.</td>
</tr>
<tr>
<td>Efficient Cryogenic Refrigeration System</td>
<td>- Cryogenic refrigeration system can reduce vehicle mass and enhance system reliability at the expense of an increased power level.</td>
</tr>
<tr>
<td>O2 - H2 ACS / RCS</td>
<td>- O2 - H2 ACS/RCS (Isp = 400 s) reduces system mass over lower Isp storables</td>
</tr>
<tr>
<td>High Isp Advanced Space Engine</td>
<td>- High Isp advanced space engine (Isp = 485 s) enhances all mission phases for all mission arch.</td>
</tr>
<tr>
<td>NTR Propulsion System</td>
<td>- NTR propulsion system for the TMI, Lunar transfer, and Mars transfer stages</td>
</tr>
<tr>
<td>Advanced In - Space Assembly Techniques</td>
<td>- Launch vehicle capability drives on - orbit assembly level.</td>
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<tr>
<td></td>
<td>- Degree of on - orbit assembly capability affects vehicle configuration, ground assembly/processing, and launch manifesting.</td>
</tr>
<tr>
<td>Advanced Materials Development</td>
<td>- Advanced materials such as metal and organic matrix composites reduce system inert mass, strength, and/or manufacturing costs.</td>
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<tr>
<td></td>
<td>- Some advanced M&amp;P may prove enabling for some mission arch. (ex: Mars/Earth capture aerobrake)</td>
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</tbody>
</table>
Schedules
Technology Development Concerns and Schedules - Cryogenic All Propulsive Vehicle

Critical technology development issues relating to the reference CAP vehicle are presented in this section. Where applicable, the same charts are also included in the CAB, NTR, NEP, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference cryogenic all propulsive vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Cryogenic Propulsion and Fluid Management
With the absence of high energy aerobraking for the all propulsive mission, cryogenic propulsion and fluid management becomes the most important technology development concern in the area of vehicle benefits. The high Isp of a LH2-LOX system (460-480 s) may prove enabling for an all propulsive mission due to the massive vehicle sizes which could result from the lower Isp (280-360 s with metallic gels) storable systems. The long term storage and low-g fluid management of cryogenic fluids, along with long lifetime, in-space restartable cryogenic engines are the major technology development concerns for a cryogenically fueled vehicle. Preliminary technology schedules are presented for space based cryogenic engines, and cryogenic fluid system development for both Lunar and Mars applications. The cryogenic space based engine development effort begins with the planned AETB work at LeRC, and continues on to development work for a large engine for Mars applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem validation tests.

Vehicle Avionics and Software
Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is produced, however. A technology development schedule for advanced communications is presented.

Life Support
A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.
Aerobraking (low energy)

Low energy aerobraking will offer mission benefits in the areas of decreased demands on the descent propulsion system, and improved crossrange capability. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems (although not as high as for high energy aerobraking), avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiative materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space assembly and ops problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule. It should be noted that this schedule was built with high energy aerobraking in mind, and will possibly be compressed to some degree if only low energy aerobraking is developed.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MEV aerobrakes. As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence.

Summary

As noted before, many of the identified critical and high leverage technology development issues are common across all of the major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H2, and possibly O2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique cryo all propulsive technology issues center around large advanced space engine advanced development. Enhancing technologies include cryogenic refrigeration (lander tanks), O2-H2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.
Preliminary SEI Technology Development Schedules

Space Based Engines

- Design & analysis methodologies for AETB engine
- Breadboard assy. & constr.
- Complete testbed-proven technology for LTV appl.
- AETB engine development (system tests)
- Component tests
- Prototype engine development
- Testbed upgrades for moderate thrust engine
- Tech. develop. complete
- High thrust cryo engine design (for MTV)
- Lunar FSD
- Mars FSD
- High thrust engine adv. development

Cryogenic Fluid Systems

- Definition Studies
- 1-g validation
- SOFTE
- LIRE, LACE
- Small scale pressure ctrl, and liquid reorient. & acq. flight tests
- Integrated subsystem breadboard demonstr.
- Initial LTV design complete
- Advanced cryo tank design for LTV
- COLD-SAT Alter. flt.
- Flight
- Analysis complete
- CFM flight experiment (-optional-COLD-SAT or alternative)
- Advanced development & flight test (program level)
- Lunar FSD
- Mars FSD
Preliminary SEI Technology Development Schedules (Cont.)

Autonomous Systems

Autonomous landing req. def.
Precision landing tech, demo. □ Hazard det. & avoidance tech. demo.
Testbed construction & operations
Precision landing sys. demo. □ Hazard det. & avoidance sys. demo.
System demonstrations (1-g)
AR&D subsystem comp. tests
GN&C & docking mech. system tests
Flight □ Cooperative AR&D flight test
Analysis
Flight □ Uncooperative AR&D flight test
Analysis

♦ Mars FSD*
♦ Lunar FSD*

* Technology should not present FSD threatening problems; current technologies adequate for minimum mission.
Preliminary SEI Technology Development Schedules (Cont.)

Life Support

Shield material data base comp ▼ ▼ GCR & SPE prediction codes within 25% accuracy
Rad shielding analytical code & materials prop. develop.
Current analysis & test cap. analysis ▼ ▼ Initial analysis & tests comp. ▼ ▼ Lunar outpost shielding concepts validation comp.
Mars vehicle shielding concepts validation
Rad. shielding materials testing & eval.

Analytical models & trade studies comp. ▼ ▼ Trade studies comp.
Regen L.S. analysis & trade studies

Lunar vehicle breadbd subsys & integrated sys ▼ ▼ Human rated RLS validation tests complete ▼ ▼ Lunar outpost food supp. system ▼ ▼ MTV Advanced RLS technologies
RLS System/subsystem tests

PLSS heat xfer & control method ▼ ▼ Surf. mobility aid requirements
EVA systems analytical studies
Mars FSD ◆

EVA ground test program Complete
Lunar EVA suit design mods for Mars surf.

- Helmet/duit display & control concepts tests
- Lunar surf suit breadbd test
- Gloves & displays in simul. SSF environment
- Regen. PLSS breadbd for lunar surf.
- Verif tests of adv. dexterious gloves & disp.
- Complete breadbd lunar EVA suit / simulated surf. cond.
## Preliminary SEI Technology Development Schedules (Cont.)

(Dates marked with these symbols: 1=2007, 2=2008, 3=2009, 4=2010)

### In-Space Assembly & Processing

| 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 | 20 |

#### LTV tech. development ground tests

- △ Design for construction guideline derivation
- △ Upgrades complete △ Testbed upgrade for advanced in space assembly & cons for adv. Lunar ops.
- △ Lab assembly of char. Mars A/B
- △ Mars A/B design for assembly

#### Ground & in-space veh processing program def.

- △ Sensors, tools, and telerob. sys for Lunar veh.
- △ Lunar veh automated test equip. breadbd demo.
- △ Breadboard construction

#### Lunar vehicle processing tests complete △ Mars vehicle processing tests complete

- △ SSF testing & operations
- △ Lunar update comp.
- △ Mars update comp.

#### Lab breadboard upgrades for surface veh. proc.

- △ Lunar FSD
- △ Mars FSD

- △ High load perm. joint breadboard
- △ Telerobotic Space welding demo.
- △ Ground lab testbed model complete (inc crane)
- △ Lunar veh utilities testbed & A/B assembly demo. complete
Preliminary SEI Technology Development Schedules (Cont.)

**Aerobraking**
- Adaptive GN&C systems development & testing
- CFD code development & analysis
- Hypersonic wind tunnel testing
- TPS materials & concepts completed
- Flight
- AFE design & materials
- AFE analysis
- Prelim. tech. dev. complete
- TPS mats. & concepts validated
- Final tech. dev. complete
- LTV & Mars telerob. aerobrake tech. dev. (program)
- Lunar FSD
- Mars FSD
- Tech dev. complete
- MTV aerobrake tech. dev.

**High Rate Communications**
- Model complete
- Deep space optical comm. experiment model development
- Component tech. development
- Critical design review
- Flight test
- Deep space optical comm. experiment (optional)
  - Lunar FSD
  - Mars FSD
  - a - Key component tech. for Ka band, TWT, and Ka band MMIC amps formulated
  - b - Automated high rate comm ops for Lunar outpost & Mars robotic demo.
Facilities
Facilities

The facility needs have only been identified in this study; the extent of the impact is yet to be determined. A "bona fide" facility development plan has not been done as some of the requirements are only at a top-level needs evaluation. Therefore, the exact nature of the subsystems and their support facilities are undetermined. When these determinations have been made for the final NASA selected vehicle, the results must be integrated with the vehicle development schedule.

In addition to the information here, additional facility and equipment detail is shown in Ground subsection of the Support Systems section of this text. The volumes for the baseline Cryo/Aerobreak vehicle for assembly, storage, and launch processing are shown in the "Facility Requirements" chart. Processing time shown in the "Assembly Time per Mission" chart. All impacts will be to increase the processing time and working volumes required. Any facility requirements must be viewed in the light of and incorporated into the National Launch Facility Plan.
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<th>Storage Volume</th>
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Costs
Cryo Aerobrake

Programmatics

The objectives of the Programmatics task during the current phase of the study were: (1) realistic initial schedules that include initial critical path program elements; (2) initial descriptions of new or unique facilities requirements; (3) development of a stable, clear, responsive work breakdown structure (WBS) and WBS dictionary; (4) initial realistic estimates of vehicle, mission and program costs, cost uncertainties, and funding profile requirements; (5) initial risk analysis, and (6) early and continuing infusion of programmatics data into other study tasks to drive requirements/design/trade decisions.

The issues addressed during the study to date included: (1) capturing all potential long-lead program items such as precursor missions, technology advancement and advanced development, related infrastructure development, support systems and new or modified facility construction, since these are as important as cost and funding in assessing goal achievability; (2) incorporating sufficient operating margin in schedules to obtain high probability of making the relatively brief Mars launch windows; (3) the work breakdown structure must support key study goals such as commonality and (4) cost estimating accuracy and uncertainty are recurring issues in concept definition studies.

Introduction

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown in "Overall Study Flow" chart. After the "neckdown" was completed, significant effort was put into programmatics.
As was indicated earlier, we established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for Lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in space transportation technologies as baselines for greater activity levels. The high level schedules developed for these three levels of activity are shown in the "Minimum Program", "Full Science Program" and "Industrialization and Settlement Program" charts and a comparison of them for both Lunar and Mars is shown in the "Lunar Program Comparison" and "Mars Program Comparison" charts. The Cryogenic-All propulsive systems were derived from the Cryo/Aerobrake systems by adjusting the size of the Trans-Mars Injection Stage and eliminating the aerobrake from the materials costed.

Schedule/Network Development Methodology

A PC system called Open Plan by WST Corporation was used, which allows direct control and lower cost over a larger (mainframe) system. The network was purposely kept simple. Summary activities were used in development of the networks. When detailed to a lower level, some activities will require a different calendar than we used. One calendar with a five day work week - no holiday was used. Utilizing multicalendars on a summary network could confuse the development. The Preliminary WBS Structure Level 7 was followed for selection of work to be detailed. An example of Level 7 is: MEV Ascent Vehicle Structure/Mechanisms. We then developed a generic logic string of activities with standard durations for like activities. This logic was then applied against each WBS Level 7 element. To establish interface ties between logic strings and determination of major events, we used the Upper Level Summary Schedule and Summary Level Technology Schedule.
Goals/Purpose

There were two goals for the schedule/network development. These were:

a. Guidelines for Future Development. The schedules are a preliminary road map to follow in the development program.

b. Layout Basis Framework for Network. The networks can be used for future detail network development. This development can be in phases retaining unattended logic for areas which can be detailed.

Status

Six preliminary networks have been developed. They are:

- Lunar minimum
- Lunar full science
- Lunar industrialization
- Mars missions
- Mars full science
- Mars settlement

These networks will be further developed as information becomes available. The technology development plan schedules are shown in the Schedules subsection of this text; an example of the standard 6 year program phase C/D schedule is shown in "Reference 6 yr. Full-Scale Development Schedule" chart. The network schedules developed during the study are available in the Final Report Costs Data Book and the WBS.

Facilities

The facility requirements and approaches are discussed in the Facilities section of this text.
Development Implementation

The integrated technology advancement and full-scale development schedules for the Cryogenic/Aerobrake is shown for the subsystems in the Schedule section of this document. The MEV is developed according to the above mentioned standard 6-year FSD schedule. The Man-rating schedules for critical systems, that must be accomplished before first flight, are given in the next six man-rating charts. The long-duration Mars Transit Habitat, and its critical subsystems, will require operational testing in space to qualify for the Mars mission. How all development and testing is actually done depends on program interrelationships between lunar and Mars missions.

Work Breakdown Structure

The approach to developing a WBS tree and dictionary was to use the Space Station Freedom Work Package One WBS as a point of departure to capture commonality, modularity and evolution potentials. We worked with MSFC to evolve the WBS illustrated in the six WBS charts given in this section. The WBS dictionary details are provided with the WBS tree in a separate deliverable document.

Cost Data

Overall Approach

Space transfer concept cost estimates were developed through parametric and detail estimating techniques using program/scenario plans and hardware and software descriptions combined with NASA and subcontractor data. Our estimating approach simulates the aerospace development and production environment. It also reflects program
options not typical of aerospace programs. This flexibility allows assessment of innovative program planning concepts.

Several tools were employed in this analysis. For developing estimates the Boeing Parametric Cost Model (PCM) designed specifically for advanced system estimating was used. It utilizes a company-wide, uniform computerized data base containing historical data compiled since 1969. The second major tool is a Boeing developed Life Cycle Cost Model. The third tool is the Boeing developed Return on Investment (ROI) Analyses.

The approach to cost estimating was to use the PCM to establish DDT&E and manufacturing cost of major hardware components or to use other estimates, (e.g. Nuclear Working Group estimator) if they were considered superior and then feed them to the LCC model. Variations on equipment hardware or mission alternatives can be run through the LCC and then compared for a return on investment. This flow is illustrated in the "Costing Methodology Flow" chart. We were able to investigate alternative concepts quickly, giving system designers more data for evolving scenario/mission responsive concepts. Transportation concepts, trade studies, and "neckdown" efforts were supported by this approach.

**Parametric Cost Model**

PCM develops cost from the subsystem level and builds upward to obtain total program cost. Costs are estimated from physical hardware descriptions (e.g., weights and complexities) and program parameters (e.g., quantities, learning curves, and integration levels). Known costs are input directly into the estimate when available; the model assesses the necessary system engineering and system test efforts needed for integration into the program. The PCM working unit is man-hours, which allows relationships that tie physical hardware descriptions first to design engineering or basic factory labor, and then through the organizational structure to pick up functional areas such as systems engineering, test, and development shop. Using man-hours instead of dollars for estimating relationships enables more reliable estimates. The PCM features, main inputs, and results are shown in the "Boeing Parametric Cost Model (PCM)" chart. The applicable PCM results, in constant 1990 dollars, are then put into the Life Cycle Cost Model to obtain cost spreads for the various missions/programs. The various hardware components
costed for the three different missions/programs are shown in the "LCCM Hardware Assignments" chart. As stated above, adjustments were made for the Cryogenic-All Propulsive from the Cryo/Aerobrake configuration.

The development of space hardware and components needed to accomplish the three different Lunar/Mars missions were identified. These components are grouped into three different categories defined below.

**HLLV (Heavy Lift Launch Vehicle)** is the booster required to lift personnel, cargo and fuels into LEO and support the LEO node operations.

**Propulsion** Includes the space propulsion system required to transfer people, cargo and equipment out of LEO and into space. Space means Lunar, Mars and Earth destinations. Propulsion Systems also include an all-propulsive cryogenic Trans Mars Injection System (TMIS) for the Minimum Mission, the Nuclear Electric Propulsion Stage for the Settlement/Industrial Missions.

**Modules** Include the space systems that are required to transfer people, cargo and equipment from LEO to Lunar and Mars orbit; to de-orbit and sustain life and operations on the Lunar and Mars Surface; and, finally, to return personnel and equipment to LEO.

**Cost Buildups**

The PCM cost Model can be used directly to obtain complete DDT&E cost, including production of major test articles, by entering into the manufacturing section the equivalent numbers of units for each item, including the first flight article. However, when operated in this way, PCM does not give the first unit cost. To save time, we operated PCM so as to give first unit cost, which we needed for life cycle cost analyses, and used the first unit cost to manually estimate the test hardware content of the DDT&E program. The "wrap factors" shown in the cost buildup sheets were derived from the PCM runs as the factor that is applied to design engineering cost to obtain complete design and development costs, e.g. including non-recurring items such as systems engineering and tooling development.

**Life Cycle Cost Model**

The LCCM cost data is a composite of HLLV costs, launch base facilities cost estimate based on $/sq. ft. and parametric estimates derived from the Parametric Cost Model. The
principal source of information is from the PCM. All hardware cost estimates, with the exception of HLLV, have been developed with this model.
The LCCM consists of three individual models. One model is for the Minimum Program Scale; the second is for the Full Science Program Scale; while the third model is for the Settlement/Industriallization Program Scale. The Minimum Program meets the President's Space Exploration Initiative (SEI) objectives. These capabilities include permanent Lunar facilities but not permanent human presence and three missions to Mars. The Full Science program not only meets the President's SEI objectives but also provides for long term bases for far-ranging surface exploration. The Settlement/Industriallization program accomplishes the objectives of the Minimum and Full Science program scales and additionally returns practical benefits to Earth. These models were developed using the three architecture levels described in the Boeing manifest worksheets. Total cost for each system are tabulated by year and each year's totals feed into a summary sheet that calculates the total program cost for each level. Since the LCCM results are mission related, not just vehicle related, they are not provided here but are available in Final Report Cost Data Book. The LCCM was developed using Microsoft Excel version 2.2 for the Macintosh computer. Any Macintosh equipped with Excel 2.2 can be used to execute the model.

Return On Investment

One of the principal uses of the LCCM is to develop trades and return on investment for technology options. As shown in the "Costing Methodology Flow" chart, two separate life cycle cost models (which include DDT&E and production cost data derived from the parametric cost models) must be developed for each ROI case; a reference, and a case utilizing a technology option. The two life cycle cost streams are separately entered, and the ROI model is executed. The flow also illustrates that not all of the data entered into the life cycle cost model is derived from available costing software. Technical analysis must accompany this data. For example, the number of units which must be produced for the DDT&E program must be determined. This is done at the subsystem level based on knowledge of past programs, and proposed system/subsystem tests. Since the ROI analysis is mission related, not just vehicle related, the data is not presented here but is available in the Final Report Costs Data Book.

Results
A summary of the cost data produced by the PCM for the CAB vehicle are given in the "Mars CAB Preliminary PCM Summary" and "Mars CAB Preliminary PCM Summary - continued" charts. The PCM program was used to produce DDT&E and production cost estimates for each of our reference Mars and lunar vehicles to the subsystem level. The DDT&E costs generated by the PCM do not include all of the necessary hardware for the first mission vehicle. Hence all necessary additional units (prototypes, test units, lab units, etc.) were added into the vehicle cost buildups as shown in the "CAB Cost Buildup" charts. The total DDT&E includes additional costs (e.g., additional units in the DDT&E program), contractor fees and the engineering wrap factor. The total DDT&E from the cost buildup and the unit cost from the PCM are the primary vehicle cost inputs to the LCC model.
Risk Analyses

Risk analyses were conducted to develop an initial risk assessment for the various architectures. This presentation of risk analysis results considers development risk, man-rating requirements, and several aspects of mission and operations risk.

Development Risk

All of the architectures and technologies investigated in this study incur some degree of development risk; none are comprised entirely of fully developed technology. Development risks are correlated directly with technological uncertainties. We identified the following principal risks:

Cryogenics - High-performance insulation systems involve a great many layers of multilayer insulation (MLI), and one or more vapor-cooled shields. Analyses and experiments have indicated the efficacy of these, but demonstration that such insulation systems can be fabricated at light weight, capable of surviving launch g and acoustics loads, remains to be accomplished. In addition, there are issues associated with propellant transfer and zero-g gauging. These, however, can be avoided for early lunar systems by proper choice of configuration and operations, e.g. the tandem-direct system recommended elsewhere in this report. This presents the opportunity to evolve these technologies with operations of initial flight systems.

Engines - There is little risk of being able to provide some sort of cryogenic engine for lunar and Mars missions. The RL-10 could be modified to serve with little risk; deep throttling of this engine has already been demonstrated on the test stand. The risk of developing more advanced engines is also minimal. An advanced development program in this area serves mainly to reduce development cost by pioneering the critical features prior to full-scale development.

Aerocapture and aerobraking - There are six potential functions, given here in approximate ascending order of development risk: aero descent and landing of crew capsules returning from the Moon, aerocapture to low Earth orbit of returning reusable lunar vehicles, landing of Mars excursion vehicles from Mars orbit, aero descent and landing of crew capsules returning from Mars, aerocapture to low Earth orbit of returning Mars vehicles, and
aero capture to Mars orbit of Mars excursion and Mars transfer vehicles. The "Development Risk for Aerobraking by Function chart provides a qualitative development risk comparison for these six functions.

Aerocapture of vehicles requires large aerobrakes. For these to be efficient, low mass per unit area is required, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials. By comparison, the crew capsules benefit much less from high performance structures and TPS.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk that has not yet been solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both. While the design challenge can probably be met, aerobrake assembly is a difficult design and development challenge, representing an important area of risk.

Nuclear thermal rockets - The basic technology of nuclear thermal rockets was developed and demonstrated during the 1960s and early 1970s. The development risk to reproduce this technology is minimal, except in testing as described below. Current studies are recommending advances in engine performance, both in specific impulse (higher reactor temperature) and in thrust-to-weight ratio (higher reactor power density). The risks in achieving these are modest inasmuch as performance targets can be adjusted to technology performance.

Reactor and engine tests during the 1960s jetted hot, slightly radioactive hydrogen directly into the atmosphere. Stricter environmental controls since that time prohibit discharge of nuclear engine effluent into the atmosphere. Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines. Full-containment facilities will be required to contain all the hydrogen effluent, presumably oxidize it to water, and remove the radioactivity.

Electric Propulsion Power Management and Thrusters - Power management and thrusters are common to any electric propulsion power source (nuclear, solar, or beamed power). Unique power management development needs for electric propulsion are (1) minimum mass and long life, (2) high power compared to space experience, i.e. megawatts instead of kilowatts, (3) fast arc suppression for protection of thrusters. Minimizing mass of power distribution leads to high distribution voltage and potential problems with plasma losses,
arcing, and EMI. Thus while power management is a mature technology, the unique requirements of electric propulsion introduce a number of development risks beyond those usually experienced in space power systems.

Electric thruster technology has been under development since the beginning of the space program. Small thrusters are now operational, such as the resistance-heat-augmented hydrazine thrusters on certain communications spacecraft. Small arc and ion thrusters are nearing operational use for satellite stationkeeping.

Space transfer demands on electric propulsion performance place a premium on high power in the jet per unit mass of electric propulsion system. This in turn places a premium on thruster efficiency; power in the jet, not electrical power, propels spaceships. Space transfer electric propulsion also requires specific impulse in the range 5000 to 10,000 seconds. Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver this performance. Ion thrusters have acceptable efficiency but relatively low power per unit of ion beam emitting area. MPD thruster technology can deliver the needed Isp with high power per thruster, but has not yet reached efficiencies of interest. Circular ion thrusters have been built up to 50 cm diameter, with spherical segment ion beam grids. These can absorb on the order of 50 kW each. A 10 MWe system would need 200 operating thrusters. The development alternatives all have significant risk: (1) Advance the state of the art of MPD thrusters to achieve high efficiency; (2) Develop propulsion systems with large numbers of thrusters and control systems; or (3) Advance the state of the art of ion thrusters to much larger size per thruster.

Nuclear power for electric propulsion - Space power reactor technology now under development (SP-100) may be adequate; needed advances are modest. Advanced power conversion systems are required to obtain power-to-mass ratios of interest. The SP-100 baseline is thermoelectric, which has no hope of meeting propulsion system performance needs. The most likely candidates are the closed Brayton (gas) cycle and the potassium Rankine (liquid/vapor) cycle. (Potassium provides the best match of liquid/vapor fluid properties to desired cycle temperatures.) Stirling cycle, thermionics, and a high-temperature thermally-driven fuel cell are possibilities. The basic technology for Brayton and Rankine cycles are mature; both are in widespread industrial use. Prototype space power Brayton and Rankine turbines have run successfully for thousands of hours in laboratories. The development risk here is that these are very complex systems; there is no experience base for coupling a space power reactor to a dynamic power conversion cycle;
there is no space power experience base at the power levels needed; and these systems, at power levels of interest for SEI space transfer application, are large enough to require in-space assembly and checkout. Space welding will be required for fluid systems assembly.

Solar power for space transfer propulsion - Solar power systems for space propulsion must attain much higher power-to-mass ratios than heretofore achieved. This implies a combination of advanced solar cells, probably multi-band-gap, and lightweight structural support systems. Required array areas are very large. Low-cost arrays, e.g. $100/watt, are necessary for affordable system costs, and automated construction of the large area structures, arrays, and power distribution systems appears also necessary. Where the nuclear electric systems are high development risk because of complexity and the lack of experience base at relevant power levels and with the space power conversion technologies, most of the solar power risk appears as technology advancement risk. If the technology advancements can be demonstrated, development risk appears moderate.

Avionics and software - Avionics and software requirements for space transfer systems are generally within the state of the art. New capability needs are mainly in the area of vehicle and subsystem health monitoring. This is in part an integration problem, but new techniques such as expert and neural systems are likely to play an important role.

An important factor in avionics and software development is that several vehicle elements having similar requirements will be developed, some concurrently. A major reduction in cost and integration risk for avionics can be achieved by advanced development of a "standard" avionics and software suite, from which all vehicle elements would depart.

Further significant cost savings are expected from advancements in software development methods and environments.

Environmental Control and Life Support (ECLS) - The main development risk in ECLS is for the Mars transfer habitat system. Other SEI space transfer systems have short enough operating durations that shuttle and Space Station Freedom ECLS system derivatives will be adequate. The Mars transfer requirement is for a highly closed physio-chemical system capable of 3 years' safe and dependable operation without resupply from Earth. The development risk arises from the necessity to demonstrate long life operation with high confidence; this may be expensive in cost and development schedule.
Man-Rating Approach

Man-rating includes three elements: (1) Design of systems to manned flight failure tolerance standards, (2) Qualification of subsystems according to normal man-rating requirements, and (3) Flight demonstration of critical performance capabilities and functions prior to placing crews at risk. Several briefing charts follow: the first summarizes a recommended approach and lists the subsystems and elements for which man-rating is needed; subsequent charts present recommended man-rating plans.

Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbital launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks.

Earth launch - The Earth launch risk to in-space transportation is the risk of losing a payload because of a launch failure. Assembly sequences are arranged to minimize the impact of a loss, and schedules include allowances for one make-up launch each mission opportunity.

Assembly and Orbital Launch Operations - Four sub-areas are covered: assembly, test and on-orbit checkout, debris, and inadvertent re-entry.

Assembly operations risk is reduced by verifying interfaces on the ground prior to launch of elements. Assembly operations equipment such as robot arms and manipulators will undergo space testing at the node to qualify critical capabilities and performance prior to initiating assembly operations on an actual vehicle.

Assembly risk varies widely with space transfer technology. Nuclear thermal rocket vehicles appear to pose minimum assembly risk; cryo/aerobraking are intermediate, and nuclear and solar electric systems pose the highest risk.

Test and on-orbit checkout must deal with consequences of test failures and equipment failures. This risk is difficult to quantify with the present state of knowledge. Indications are: (1) large space transfer systems will experience several failures or anomalies per day. Dealing with failures and anomalies must be a routine, not exceptional, part of the operations or the operations will not be able to launch space transfer systems from orbit; (2)
vehicles must have highly capable self-test systems and must be designed for repair, remove and replace by robotics where possible and for ease of repair by people where robotics cannot do the job; (3) test and on-orbit checkout will run concurrently with propellant loading and launch countdowns. These cannot take place on Space Station Freedom. Since the most difficult part of the assembly, test and checkout job must take place off Space Station Freedom the rest of the job probably should also.

Orbital debris presents risk to on-orbit operations. Probabilities of collision are large for SEI-class space transfer systems in low Earth orbit for typical durations of a year or more. Shielding is mandatory. The shielding should be designed to be removed before orbital launch and used again on the next assembly project.

Creation of debris must also be dealt with. This means that (1) debris shielding should be designed to minimize creation of additional debris, especially particles of dangerous size, and (2) operations need to be rigorously controlled to prevent an inadvertent loss of tools and equipment that will become a debris hazard.

Inadvertent re-entry is a low but possible risk. Some of the systems, especially electric propulsion systems, can have very low ballistic coefficient and therefore rapid orbital decay rate. Any of the SEI space transfer systems will have moderately low ballistic coefficient when not loaded with propellant. While design details are not far enough along to make a quantitative assessment, parts of these vehicles would probably survive reentry to become ground impact hazards in case of inadvertent reentry. For nuclear systems, it will be necessary to provide special support systems and infrastructure to drive the probability of inadvertent reentry to extremely low levels.

Launch Windows - Launch windows for single-burn high-thrust departures from low Earth orbit are no more than a few days because regression of the parking orbit line of nodes causes relatively rapid misalignment of the orbit plane and departure vector. For lunar missions, windows recur at about 9-day intervals.

For Mars, the recurrence is less frequent, and the interplanetary window only lasts 30 to 60 days. It is important to enable Mars launch from orbit during the entire interplanetary window. Three-impulse Mars departures make this possible; a plane change at apogee of the intermediate parking orbit provides alignment with the departure vector. Further
analysis of the three-burn scheme is needed to assess penalties and identify circumstances where it does not work.

Launch window problems are generally minimal for low-thrust (electric propulsion) systems.

Mission Risk - Comparative mission risk was analyzed by building risk trees and performing semi-quantitative analysis. The next chart presents a comparison of several mission modes; after that are the risk trees for these modes.

Ionizing Radiations and Zero G - The threat from ionizing radiations is presented elsewhere in this document. Presented here are the mitigating strategies for ionizing radiations and zero g.

Nuclear systems operations present little risk to flight crews. Studies by University of Texas at Austin showed that radiation dose to a space station crew from departing nuclear vehicles is very small provided that sensible launch and flight strategies are used. On-board crews are protected by suitable shielding and by arrangement of the vehicle, i.e. hardware and propellant between reactors and the crew and adequate separation distances. After nuclear engines are shut off, radiation levels drop rapidly so that maneuvers such as departure or return of a Mars excursion vehicle are not a problem. On-orbit operations around a returned nuclear vehicle are deferred until a month or two after shutdown, by which time radioactivity of the engine is greatly reduced.

Reactor disposal has not been completely studied. Options include solar system escape and parking in stable heliocentric orbits between Earth and Venus.

Crew radiation dose abatement employs "storm shelters" for solar flares, and either added shielding of the entire vehicle or fast transfers (or both) to reduce galactic cosmic ray exposure. Assessments are in progress; tradeoffs of shielding versus fast trips have yet to be completed. Expected impact for lunar missions is negligible and for Mars missions, modest.
Lunar Program Comparison

Lunar Population

Cumulative Delivered Payload, t.

Program Year

2000 2010 2020

Industrialization

Full Science

Minimum

Cargo

Program Year

2000 2010 2020

Industrialization

Full Science

Minimum

Flight Rate

Annual Lunar Trips (crew & cargo)

Program Year

2000 2010 2020

Industrialization

Full Science

Minimum
Mars Program Comparisons

**Population**
- Minimum/Full Science
- Settlement

**Cargo**
- Full Science
- Minimum
- Settlement

**Flights**
- Full Science
- Minimum
- Settlement
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<td>96-00</td>
<td>AFE flight: lunar return conditions including wake heating and radiation</td>
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<td>01-05</td>
<td>Plasma tunnel tests of TPS and joint leakage tolerance</td>
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<td>06-10</td>
<td>Prototype lunar brake assembly test, Shuttle</td>
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<td>11-15</td>
<td>Tandem LTV booster recovery, lunar aerobrake demo</td>
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<td>16-20</td>
<td>Manned lunar aerobrake first flight</td>
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<td>21-25</td>
<td>Mars aeronomy orbiter; Mars atmosphere statistics</td>
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<td>Mars robotic precursor using aerobrake</td>
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<td>Mars aerobrake assembly demo at SSF</td>
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<td>Mars cargo landing; aerocapture &amp; landing demo</td>
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<td>Manned Mars mission using aerobraking</td>
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<td>91</td>
<td>Ground test demonstration of 30K engine cycle for high Isp &amp; deep throttling</td>
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<td>Ground test demonstration of 150K-class TMIS engine cycle for high Isp</td>
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<td>Lunar mission performance of LTV engine</td>
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<td>Flight demo of transfer and gauging systems</td>
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<td>98</td>
<td>Qualification of Mars TMIS engine (cryo aerobraking or cryo all-propulsive architectures)</td>
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<tr>
<td>99</td>
<td>Manned Mars mission with TMIS engine</td>
</tr>
<tr>
<td>Year</td>
<td>Event</td>
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<tr>
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<td>1991</td>
<td>Ground test demonstration of lightweight high-performance insulation systems</td>
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<tr>
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<td>Ground test demonstration of insulation systems integrity for launch environment</td>
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<tr>
<td>1994</td>
<td>Early shuttle flight tests of zero-g fluids with cryo simulants, e.g. freon</td>
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<td>Ground test demonstration; selection of 0-g transfer &amp; gauging techniques; systems</td>
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<td>1997</td>
<td>Qualification of tandem LTV cryo system (no transfer or 0-g gauging)</td>
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<td>Flight measurements of LTV cryo system performance (system first flight)</td>
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<td>2004</td>
<td>Flight demo of insulation systems at Mars mission performance levels</td>
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<td>2005</td>
<td>Manned Mars mission with cryo propellants</td>
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## Advanced Auxiliary Propulsion Man-Rating Approach

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<td>Manned Mars mission with TMIS engine</td>
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* If current technology is selected, no advancement activity is required.
<table>
<thead>
<tr>
<th>Year</th>
<th>Event</th>
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<tbody>
<tr>
<td>1991</td>
<td>Technology demo of advanced components, e.g., hexad</td>
</tr>
<tr>
<td>1992</td>
<td>Brassboard demo of standard avionics architecture &amp; building blocks</td>
</tr>
<tr>
<td>1993</td>
<td>Software development environment ready</td>
</tr>
<tr>
<td>1994</td>
<td>SIL ready</td>
</tr>
<tr>
<td>1995</td>
<td>Lunar avionics qualified</td>
</tr>
<tr>
<td>1996</td>
<td>First lunar mission</td>
</tr>
<tr>
<td>1997</td>
<td>Mars robotic precursor using aerobrake; GN&amp;C demo</td>
</tr>
<tr>
<td>1998</td>
<td>Mars avionics lifetime &amp; redundancy mgmt demo (includes vehicle health monitoring &amp; onboard maintenance)</td>
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<tr>
<td>1999</td>
<td>Mars avionics qualified by ground test</td>
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<tr>
<td>2000</td>
<td>Mars cargo landing; aerocapture &amp; landing GN&amp;C demo</td>
</tr>
<tr>
<td>2001</td>
<td>Manned Mars mission</td>
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</table>
# ECLSS Systems Man-Rating Approach

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<tr>
<th>Year</th>
<th>Event</th>
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<tbody>
<tr>
<td>91</td>
<td>CELSS technology baseline selected</td>
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<tr>
<td>92</td>
<td>Qualification of Space Station Freedom ECLSS (adequate for early lunar missions)</td>
</tr>
<tr>
<td>93</td>
<td>Integrated CELSS brassboard system ground-based demonstration</td>
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<tr>
<td>94</td>
<td>Selection of Mars ECLSS technologies</td>
</tr>
<tr>
<td>95</td>
<td>Mars ECLSS begin full-scale development</td>
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<tr>
<td>96</td>
<td>Early CELSS experiments on Space Station Freedom</td>
</tr>
<tr>
<td>97</td>
<td>CELSS experiments on Space Station Freedom pocket lab</td>
</tr>
<tr>
<td>98</td>
<td>Begin experimental CELSS use on Moon</td>
</tr>
<tr>
<td>99</td>
<td>Ground qualification and life tests of Mars ECLSS complete</td>
</tr>
<tr>
<td>00</td>
<td>Zero-g qualification of Mars ECLSS on Space Station Freedom (pocket lab)</td>
</tr>
<tr>
<td>01</td>
<td>First manned Mars mission</td>
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<tr>
<td>02</td>
<td>Begin operational CELSS use on Moon</td>
</tr>
<tr>
<td>03</td>
<td>Mars CELSS qualified by ground test</td>
</tr>
<tr>
<td>04</td>
<td>Mars CELSS qualified by lunar use</td>
</tr>
</tbody>
</table>
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS

1.1 LUNAR EXPLORATION & HABITATION
1.1.1 NEW TECHNOLOGY
1.1.2 ADVANCED DEVELOPMENT
1.1.3 PRECURSOR SYSTEMS
1.1.4 SURFACE SYSTEMS
1.1.5 SURFACE MOBILITY
1.1.6 SPACE STATION ACCOMMODATIONS
1.1.7 EARTH TO ORBIT SYSTEMS
1.1.8 PAYLOAD & SCIENCES
1.1.9 MARS TRANSFER SYSTEMS (MISSION VEHICLE)
1.1.10 MANAGEMENT & INTEGRATION
1.2 MARS EXCURSION VEHICLE
1.3 MARS EXCURSION VEHICLE
1.4 ABOARD SUPPORT EQUIPMENT
1.4.1 DESIGN INTEGRATION
1.4.2 STRUCTURES/Mechanisms
1.4.3 AVIONICS
1.4.4 ELECTRICAL POWER
1.4.5 SOFTWARE
1.4.6 MANUFACTURING & ASSEMBLY
1.4.7 TEST ARTICLE INTEGRATION, ASSEMBLY & TEST
1.5 MARS TRANSFER VEHICLE
1.6 MANAGEMENT & INTEGRATION

(SEE PAGE 3)
(SEE PAGE 6)
Costing Methodology Flow

Program A
- Reference -

Element Technical Data & Analysis
Architecture Commonality
Program Scenarios

Assessment of hardware consumed & produced by DDT&E program

Element Development (DDT&E) & unit cost estimates from PCM (where practical)

Major cost of facilities estimates (test & launch)

Annual funding estimates derived from spread functions

Program B
- Option -

Element Technical Data & Analysis
Architecture Commonality
Program Scenarios

Assessment of hardware consumed & produced by DDT&E program

Element Development (DDT&E) & unit cost estimates from PCM (where practical)

Major cost of facilities estimates (test & launch)

Annual funding estimates derived from spread functions

Element development and buy schedules

Return on Investment (ROI) Analysis
ROI result for technology option vs. reference
# Boeing Parametric Cost Model (PCM)

## Features

- Designed specifically for advanced system estimating
- Uses company-wide, uniform computerized data base
- Contains historical data compiled since 1969
- Allows direct input of known costs into the estimate

## Main Inputs

<table>
<thead>
<tr>
<th>Main Inputs</th>
<th>Results</th>
</tr>
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<tbody>
<tr>
<td><strong>Hardware Characteristics</strong></td>
<td><strong>DDT&amp;E and Manufacturing Estimates</strong></td>
</tr>
<tr>
<td>- Category (e.g., primary structure, power conditioning, etc.)</td>
<td>- Based on previous Boeing programs</td>
</tr>
<tr>
<td>- Weight (or Thrust)</td>
<td>- Provides first flight unit costs</td>
</tr>
<tr>
<td>- Complexity</td>
<td>- Excludes test hardware</td>
</tr>
<tr>
<td>- % Off-the-Shelf</td>
<td>- Excludes fees</td>
</tr>
<tr>
<td>- Maturity</td>
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<tr>
<td>- Quantity</td>
<td><strong>New hardware must be relatable to PCM</strong></td>
</tr>
<tr>
<td>- Manufacturing Learning Curve</td>
<td>database to produce reasonable estimate</td>
</tr>
<tr>
<td><strong>Support Cost Factors</strong></td>
<td><strong>PCM estimates improve with increasing</strong></td>
</tr>
<tr>
<td>- Systems Engineering</td>
<td><strong>hardware detail.</strong></td>
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<tr>
<td>- Management</td>
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<tr>
<td>- Operations</td>
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<td>- Spares</td>
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### LCCM Hardware Assignments

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# Mars CAB Preliminary PCM Summary

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<th>Manufacturing ($Millions)</th>
<th>Total ($Millions)</th>
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<td>Engines</td>
<td>710.078</td>
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<td>Mars Transfer Stage</td>
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<td>Engines</td>
<td>225.780</td>
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<td>MTV Aerobrake</td>
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<td>Mars Transfer Crew Module</td>
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<td>Science</td>
<td>100.651</td>
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<td>Mars Excursion Stage</td>
<td>66.817</td>
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<td>Aeroshell</td>
<td>112.107</td>
<td>56.312</td>
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<td>Mars Excursion Vehicle Crew Cab</td>
<td>142.155</td>
<td>110.413</td>
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<td>Modified Crew Return Vehicle</td>
<td>279.935</td>
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<td>Hardware Final Ass'y and C/O</td>
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<td>Spares</td>
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<td>Hardware Total Costs</td>
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<td>System Engineering &amp; Integration</td>
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<td>Systems Flight Test Conduct</td>
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<tr>
<td>Peculiar Support Equipment</td>
<td>979.369</td>
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<td>1105.204</td>
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Mars CAB Preliminary PCM Summary - continued

Preliminary PCM Hardware Cost Estimates for CAB

- Trans Mars Injection Stage: 30.87%
- Engines: 14.50%
- Mars Transfer Stage: 9.21%
- Engines: 8.60%
- MTV Aerobrake: 8.25%
- Mars Transfer Crew Module: 7.20%
- Science: 3.01%
- Mars Excursion Stage: 2.92%
- Aerotail: 4.00%
- Mars Excursion Vehicle Crew Cab: 3.33%
- Modified Crew Return Vehicle: 3.33%
- Hardware Final Assembly and C/O: 2.79%
- Spares: 2.00%

Preliminary PCM Support Effort Cost Estimates for CAB

- System Engineering & Integ: 37.87%
- Software Engineering: 26.08%
- Systems Ground Test Conduct: 10.72%
- Peculiar Support Equipment: 9.61%
- Tooling & Special Test Equip: 7.86%
- Test Direct Quality Assurance: 6.58%
- Logistics: 3.33%
- Linkon Engineering: 2.20%
- Data: 2.79%
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CAB Cost buildup
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Cost Estimation Ground Rules

- Parametric for Hardware Elements
- Each Major Elements = Developmental Project
- ETFU for Developmental Program
- Cost/Mass Identification/Parametricization
- No Contingencies (NASA to provide)
- Payload Costs Factored
- No Learning Curve for < 4 units/yr
- 15% Initial Spares + 10% of Active Mass/yr of Service for Reusable or Long Life
- Mission Ops Support Factored Only
- Ground Ops Factored From Hardware Cost
- Lumped Cost Spread (eg DD T&E)
- SE&I & Management Costs Factored
- In Space Support Factored
- Limited To Through STV Integration (eg SE&I)
  - No Mission
  - No Carry on
  - No Launch Vehicles (will use estimates for necessary cost trades)

- PRICE - S for Software
- PRICE - H for Hardware
- Ground Support - $/FT$
Cost Analyses

Preliminary Work on:

- MMV DDT & E
- MMV Manufacturing
- MMV Support (Manufacturing & DDT & E)
- ECCV Unit Manufacturing by Subsystem
- TMIS/MTV Manufacturing by Subsystem
- MEV Manufacturing by Subsystem

First Release Scheduled for May 30, 1990